

NORTH ATLANTIC TREATY ORGANIZATION



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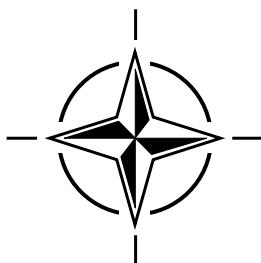
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RTO LECTURE SERIES 218

Aging Engines, Avionics, Subsystems and Helicopters

(Moteurs, avionique, sous-systèmes et hélicoptères de
générations précédentes)

The material in this publication was assembled to support a Lecture Series under the sponsorship of the Applied Vehicle Technology Panel (AVT) and the Consultant and Exchange Programme of RTO presented on 23-24 October 2000 in Atlantic City, USA and 26-27 October 2000 in Madrid, Spain.



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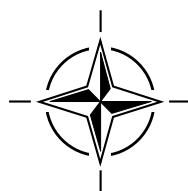
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The Research and Technology Organization (RTO) of NATO

RTO is the single focus in NATO for Defence Research and Technology activities. Its mission is to conduct and promote cooperative research and information exchange. The objective is to support the development and effective use of national defence research and technology and to meet the military needs of the Alliance, to maintain a technological lead, and to provide advice to NATO and national decision makers. The RTO performs its mission with the support of an extensive network of national experts. It also ensures effective coordination with other NATO bodies involved in R&T activities.

RTO reports both to the Military Committee of NATO and to the Conference of National Armament Directors. It comprises a Research and Technology Board (RTB) as the highest level of national representation and the Research and Technology Agency (RTA), a dedicated staff with its headquarters in Neuilly, near Paris, France. In order to facilitate contacts with the military users and other NATO activities, a small part of the RTA staff is located in NATO Headquarters in Brussels. The Brussels staff also coordinates RTO's cooperation with nations in Middle and Eastern Europe, to which RTO attaches particular importance especially as working together in the field of research is one of the more promising areas of initial cooperation.

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- SCI Systems Concepts and Integration
- SET Sensors and Electronics Technology
- IST Information Systems Technology
- AVT Applied Vehicle Technology
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Aging Engines, Avionics, Subsystems and Helicopters

(RTO EN-14)

Executive Summary

Aging aircraft concerns have dramatically escalated during recent years. Due to NATO's changing role, which includes peace keeping missions remote from home bases, the requirement of unimpaired high operational capacity, higher utilization of its air fleets and budgetary constraints, prospects are for aging aircraft problems to continue to become more acute. Airworthiness concerns have preoccupied authorities wrestling with the aging aircraft issue. However, the NATO nations are having to contend with another dimension which could not at all have been anticipated even some years ago, viz., severe budgetary constraints have necessitated retention of aircraft designed decades ago for much longer, meaning that the performance of such aircraft have to satisfy fast-changing mission needs, which might be considerably beyond their original design basis. On account of the currency and interest in the subject a Lecture Series on Aging Aircraft is proposed.

A Lecture Series (LS-206) titled "Aging Combat Aircraft Fleets – Long Term Implications," under the sponsorship of AGARD was offered in 1996. Due to the immediacy of structural airworthiness concerns, LS-206 largely dealt with issues pertaining to forms of structural degradation in aging aircraft. Also, due to the vast scope of concerns relating to the airframe in an aging aircraft, the coverage of the previously offered LS virtually excluded other structures and subsystems. For instance, it did not address any aspect of aging helicopters – a major component of the defense force of the NATO alliance. The LS was even devoid of guidance as to how to deal with other critical subsystems in aging airplanes such as avionics and electrical related subsystems and aircraft engines. Yet a recent study indicates, for instance, that the power plant and ancillary components account for some 30% of the life-cycle-cost of an aircraft. One of the principal aims of the proposed LS is to highlight schemes for retrofitting major subsystems, other than the airframe, in aging aircraft. Performance enhancement of subsystems to counter their technological obsolescence will be the major theme of the proposed LS.

The material in this publication was assembled to support a Lecture Series under the sponsorship of the Applied Vehicle Technology Panel (AVT) and the Consultant and Exchange Programme of RTO presented on 23-24 October 2000 in Atlantic City, USA and on 26-27 October 2000 in Madrid, Spain.

Moteurs, avionique, sous-systèmes et hélicoptères de générations précédentes

(RTO EN-14)

Synthèse

Le problème des aéronefs vieillissants s'est considérablement amplifié ces dernières années. Etant donné l'évolution du rôle de l'OTAN, qui comprend désormais des missions de maintien de la paix à distance des bases d'attache, la nécessité d'une grande disponibilité opérationnelle, et l'utilisation croissante des flottes aériennes dans une période de contraintes budgétaires, il est fort probable que ce problème s'accentue à l'avenir. La question de l'aptitude au vol préoccupe les autorités qui cherchent une solution au problème des aéronefs vieillissants. Toutefois, les pays membres de l'OTAN doivent aujourd'hui faire face à un autre aspect du problème, qui ne pouvait être anticipé il y a quelques années: d'importantes restrictions budgétaires nécessitant le maintien pour encore quelques temps d'appareils conçus il y a des dizaines d'années, les performances de ces derniers doivent répondre à l'évolution rapide de la nature de la mission, laquelle pourrait largement dépasser le cadre de sa conception initiale. L'actualité du sujet et l'intérêt qui lui est porté conduisent à proposer un cycle de conférences sur les aéronefs vieillissants.

Un cycle de conférences (LS-206), intitulé « Le vieillissement des flottes d'avions de combat – Implications à long terme » et organisé sous l'égide d'AGARD, a été proposé en 1996. Du fait de l'urgence des préoccupations d'aptitude au vol du point de vue structural, le cycle de conférences a essentiellement traité de points relatifs aux types de dégradation structurelle d'aéronefs vieillissants. De même, étant donné le grand nombre de questions que soulève la cellule d'un appareil vieillissant, les autres structures et sous-systèmes n'ont pratiquement pas été abordés par le cycle de conférences précédent. Ainsi, le cycle n'a évoqué aucun aspect relatif aux hélicoptères vieillissants, composante majeure de la force de défense de l'Alliance. Il n'a pas non plus fourni de ligne directrice permettant de gérer d'autres sous-systèmes essentiels sur les aéronefs vieillissants comme l'avionique, les sous-systèmes électriques et les moteurs. Pourtant, une étude récente a montré à titre d'exemple que le groupe propulseur et les éléments annexes représentent environ 30% du coût global de possession d'un aéronef. L'un des principaux objectifs du cycle de conférences proposé est de mettre en évidence les programmes de rattrapage des sous-systèmes majeurs (autres que la cellule) sur les appareils vieillissants. L'accroissement des performances des sous-systèmes visant à pallier leur obsolescence technologique constituera le thème majeur du cycle de conférences proposé.

Cette publication a été rédigée pour servir de support de cours pour le Cycle de conférences 218, organisé par la Commission des technologies appliquées aux véhicules (AVT) dans le cadre du programme des consultants et des échanges de la RTO du 23-24 octobre 2000, à Atlantic City, Etats-Unis, et du 26-27 octobre 2000 à Madrid, Espagne.

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† Paper not available at time of printing.

Theme

Aging Aircraft concerns have dramatically escalated in the military community and commercial aviation during the past decade. The percentage of aircraft, operated beyond their original design life is steadily increasing. Some models, which have already been in service for more than 40 years, will need to be retained for another two decades or longer, often serving in roles and in theaters very different from what was envisioned when they were originally designed.

Aging Aircraft has several connotations. To name a few: technological obsolescence, the specter of runaway maintenance costs, and safety. However, the adverse impact on sustainment of the fleet is the common thread.

There are other considerations when dealing with the Aging Aircraft issue; for example, spare parts, processes and tooling may no longer be available, logistic procedures may have changed and suppliers may be out of the business. Budgetary limitations and higher fleet utilization will increase the demand to cope with aging structures and major subsystems like engines and avionics. Heightening the awareness in the user community about typical challenges and technical solutions to ameliorate some of the concerns is the purpose of this Lecture Series.

Thème

Le problème du vieillissement des aéronefs s'est considérablement amplifié pour les exploitants militaires et commerciaux au cours de la dernière décennie. Le pourcentage d'aéronefs en exploitation au-delà de leur durée de vie théorique augmente régulièrement. Certains modèles, déjà en service depuis plus de 40 ans, devront être maintenus pendant encore deux décennies au moins, souvent pour des missions et des théâtres très différents de ceux qui étaient envisagés à l'origine.

Le terme “aéronefs vieillissants” a plusieurs connotations différentes, parmi lesquelles l'on peut distinguer l’obsolescence technologique, le spectre des coûts de maintenance incontrôlés et les considérations de sécurité. Mais, tous ces aspects ont un facteur commun: l'impact négatif sur le maintien de la flotte.

Il y a aussi d'autres considérations à prendre en compte; par exemple la disponibilité de pièces de rechange, de processus et d'outillage, les procédures logistiques qui peuvent avoir changé, et les fournisseurs qui peuvent avoir fait faillite. Les limitations budgétaires et l'utilisation accrue des flottes aériennes nécessiteront de porter plus d'attention aux aspects de vieillissement de la structure et des sous-systèmes principaux des aéronefs, tels que les moteurs et l'avionique. Ce cycle de conférences a pour objectif de promouvoir une meilleure sensibilisation des utilisateurs aux défis et aux solutions techniques typiques susceptibles de pallier à certains de ces problèmes.

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J85 REJUVENATION THROUGH TECHNOLOGY INSERTION

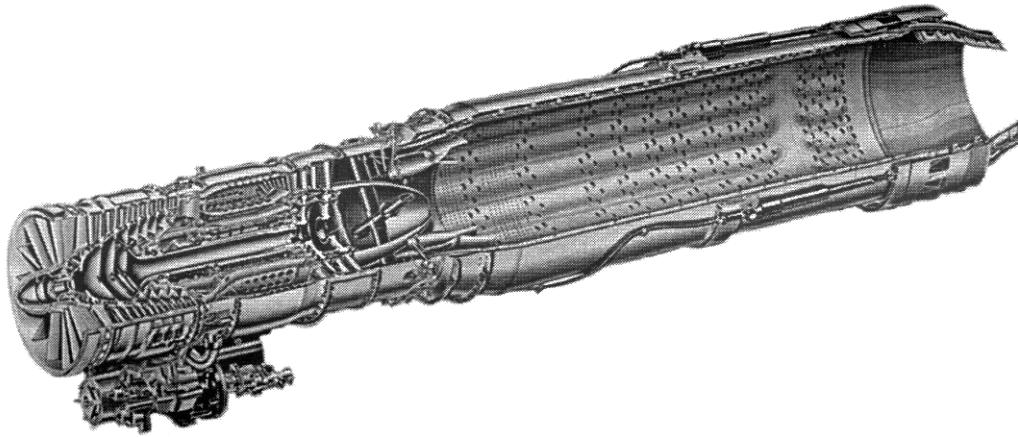
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Military Engines Operation

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Summary

The history of the General Electric J85 turbojet engine is presented from its early inception as a single-use drone engine to its equipping of almost 3,000 frontline fighter and attack aircraft. The technology development to enable this progression and provide support of over 7,000 in-service engines is further discussed, allowing its continued use for the next 40 years. Specific examples of technology insertion are detailed, including a discussion of the innovative program leading to upgrade of the J85-powered T-38 fleet and options for the rest of the world's air forces.

thrust to weight ratio turbojet engines with potential application to early cruise missiles and drones. Initially using a compressor with only seven stages, a configuration using eight stages was finally settled on to provide for adequate performance margin, which proved to be prophetic. These design efforts resulted in the J85-GE-1 engine, and at 1,900-2,100 lbs. of dry thrust powered the GAM-72/ADM-20 Quail decoy missile (Figure 1) deployed on USAF B-52 bombers. Besides needing high thrust in a lightweight package, most other requirements were fairly benign. A reasonable astart envelope was specified due to its being an air-launched missile, and a 30-minute, one-way mission life was all that was needed.

Keywords

J85, CJ610, CF700, T-38, F-5, PMP, turbojet, compressor, spooled rotor, Inconel, ejector

Initial J85 Development

In the early 1950s, General Electric (GE) began design work on a series of very small, high

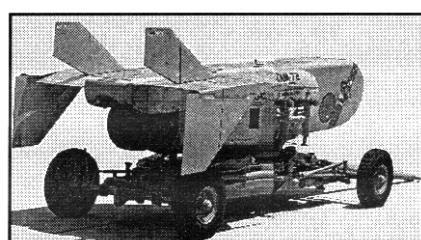


Figure 1: McDonnell Douglas ADM-20 (GAM-72) Quail

The high thrust to weight ratio, small size, and good fuel economy brought the engine to the attention of aircraft designers at Northrop. These designers were in the process of digesting the lessons of the air war in Korea, where high speed, high ceiling, maneuverability, and long range were all at a premium. Northrop was also analyzing the results of a survey they conducted with users around the world, which drove Northrop towards development of a high-performance lightweight fighter (World Airpower Journal, Vol. 25, Summer 1996).

The J85 goes Manned and Reusable

By 1954, Northrop's aircraft studies in lightweight fighters and associated trainers centered on using the J85 (Scutts, 1986). Their N-156 series of designs utilized two of the engines in an afterburning configuration to achieve the necessary maximum speed and maneuvering capability. It was at this point that the humble beginnings of the J85 as a missile engine began to raise problems.

Selected by the USAF to build a supersonic jet trainer, Northrop's N-156T design became the YT-38 Talon. This selection was dependent on successful flight testing of prototype aircraft powered by J85 engines (Scutts). However, design work was necessary to increase the dry thrust of the engine, increase its service life, and adapt it to withstand the effects of the afterburner, not to mention a greatly expanded flight envelope. Pending the results of these efforts, however, the decision was made to install modified, non-afterburning J85-GE-1 engines in the first prototype T-38. April 10, 1959 marked the first flight of the J85 as the primary powerplant of a manned aircraft, successfully reaching 0.9M and a height of 35,000 feet, achieving supersonic flight in a shallow dive only a few days later (Scutts). The second prototype was also outfitted in this manner.

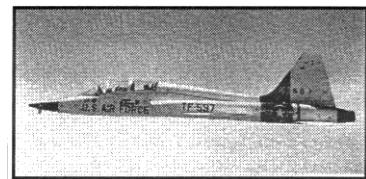


Figure 2: Northrop T-38A

Following successful development, the definitive afterburning version of the J85 for T-38 aircraft (Figure 2) was the J85-GE-5, which was installed on the first production aircraft and began flight testing in January 1960. The -5 produced 2,850 lbs. of thrust dry, and 3,850 lbs. of thrust in afterburner. In all, 1,187 T-38s were produced.

Closely following the development of the T-38 was Northrop's N-156F design, a single-seat fighter sharing many of the major components and characteristics of the T-38 and eventually becoming the F-5 (Figure 3). Being so close in timing to the T-38, the first F-5 prototype was also fitted with J85-GE-1s, going supersonic on its July 30, 1959 maiden flight even with these non-afterburning engines (Scutts). The production F-5A/B series of aircraft were equipped with the J85-GE-13, producing 4,080 lbs. in afterburner.

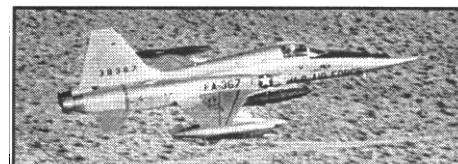


Figure 3: Northrop F-5A

The material used in the eight-stage compressor rotor and blades was steel-based AM355. These materials were consistent with the expected usage, environment, and anticipated service life of these original aircraft and other potential applications. The inherent versatility and flexibility of the J85 led to the development of multiple versions to be the primary powerplant for a substantial number of military aircraft, and as an auxiliary thrust powerplant for others (Figure 4). It even came full circle back to its beginnings, powering the Teledyne Ryan MQM-34 Mod II reusable drone.

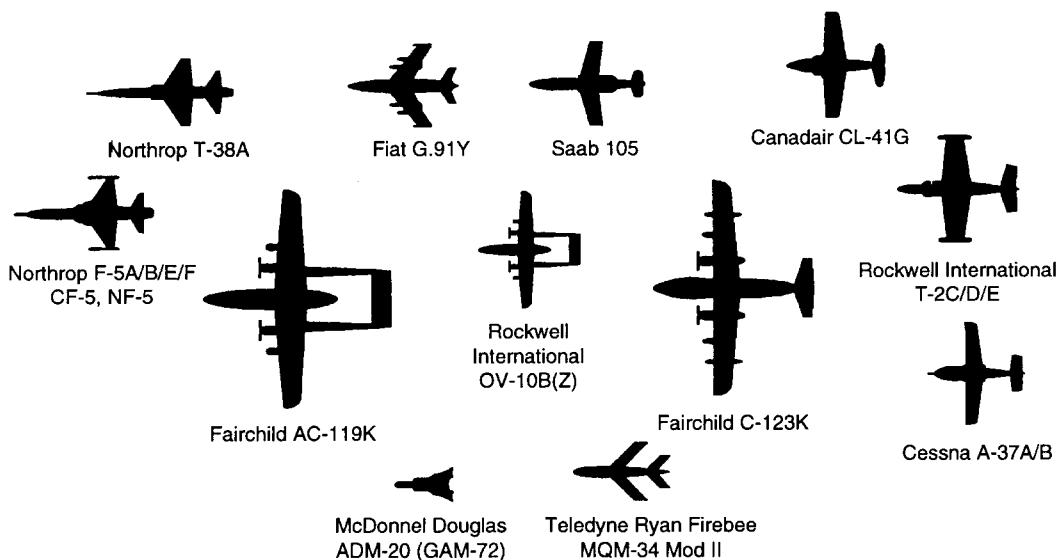


Figure 4: The J85 Aircraft Family

Being incorporated on this wide variety of aircraft ensured a long-lived, robust production volume. An impressive total of 9,592 engines were built, including licensed production, as shown in Table 1. At last count, some 4,500 of these engines were still in active service around the world, accumulating over 900,000 flight hours annually. In addition to military production, the J85 was

developed into civil versions as well. The CJ610 and CF700 series of engines shared the compressor and basic configuration of the non-afterburning J85 versions, with the CF700 utilizing an aft fan for increased performance. Over 3,100 of these engines were produced and powered the early Learjet 24/25, Hansa, Westwind, Falcon 20, and Sabre 75 business jets.

Model	Number Produced	Aircraft Type(s)	Engine Type	Thrust (lbs)
J85-GE-4	740	T-2C/D/E; OV-10B(Z)	Dry	2,950
J85-GE-5	2,826	T-38	Afterburning	3,850
J85-GE-7	577	ADM-20, MQM-34	Dry	2,100
J85-GE-13	2,330	F-5A/B	Afterburning	4,080
J85-GE-13A*	201	G.91Y	Afterburning	4,080
J85-CAN-15*	609	CF-5, NF-5	Afterburning	4,300
J85-GE-17	544	AC-119K, C-123K	Dry	2,850
J85-GE-17A	1,404	A-37	Dry	2,850
J85-GE-17B	101	Saab 105	Dry	2,850
J85-CAN-40*	260	CL-41	Dry	2,850
Total	9,592			
* = Licensed Production				

Table 1: Eight-Stage Compressor J85 Engine Production

A New Nine-Stage Compressor for Increased Thrust

As with most fighter aircraft programs, Northrop began investigating ways of offering improved performance in its F-5 series to counter the growing capabilities of threat aircraft and countries. Predictably, these centered on increasing the thrust available from the current J85 series. GE had begun testing on a new nine-stage compressor of increased diameter in 1963 (Scutts), with the goal of creating a J85 that could produce over 3,800 lbs. of thrust dry and 5,000 lbs. of thrust in afterburner. The new compressor rotor and blades were made of titanium alloy, primarily to decrease the weight associated with the increased size. The rotor itself eliminated many of the individual disks that were used in the previous compressor by going to a spool concept, which essentially combined multiple disks into a single continuous part. These changes produced a compressor with a much greater pressure ratio and airflow, decreased parts count, and which also drastically reduced any susceptibility to corrosion compared to the previous AM355-based eight-stage compressor.

This version of the J85 became the J85-GE-21, and was flown on a modified F-5B on March 28, 1969. The -21 became the production engine for the F-5E/F aircraft, and flew for the first time on an F-5E (Figure 5) in August 1972 (Scutts). When production of this engine ceased in 1987, 3,482 units had been produced including 170 under license. Approximately 2,500 engines remain in service today.

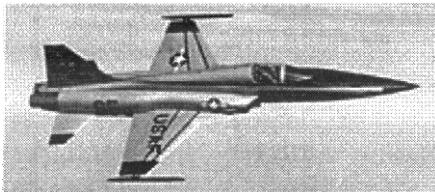


Figure 5: Northrop F-5E

Updating the Eight-Stage Compressor

In the 1990s, the USAF had approximately 500 T-38s powered by the J85-GE-5 still in service, with other users having active fleets as well. Various studies and analyses by the USAF on how best to accommodate their advanced flight training requirements led to the development of a

comprehensive T-38 upgrade program. This program committed substantial resources to avionics and aircraft structural improvements significant enough to warrant an aircraft designation change (to T-38C), enabling T-38 operations until the 2040 timeframe, some eighty years after its first flight. However, limited effort was devoted to addressing propulsion system reliability and performance. It was against this backdrop that the 1995 loss of a T-38 due to engine failure led to efforts to address modernization of the propulsion system as well.

Through operational experience, it became known that the AM355 material properties and disk configurations of the eight-stage compressor were susceptible to corrosion over time, which led to pits and cracks affecting the calculated low-cycle fatigue limits. A series of life limit decreases were implemented over the ensuing years, driving increased maintenance costs of the J85 to ensure adequate safety. Exacerbating this susceptibility were changes in the mission environment and usage that were not fully documented. Understandably, subtleties in the usage and its effects on the design had crept in since the original rotor was developed in the late 1950s, not the least of which was a 50% increase in the mission severity factor for low-cycle fatigue. The management and premature retirement of the compressor rotors became the number one maintenance driver for the T-38 as a result.

Development of a spooled rotor, similar to the design of the nine-stage -21 titanium compressor, was proposed through the USAF Component Improvement Program (CIP) in 1994 to address the reduced compressor rotor life limits. The cost of this program exceeded available USAF funding at the time, and the life cycle cost analysis did not support giving the project very high priority. In 1995, operational events drove a re-examination of the need for a new rotor, with the most significant one being the loss of a T-38 due to engine failure. Investigation into this mishap revealed that an uncontained failure of an 8th stage compressor disk had occurred, with corrosion identified as the root cause. Clearly, a program to address the life of the compressor rotor was required if the T-38 were to remain operational as planned.

The USAF's boundary conditions for developing a new rotor system for the J85-GE-5 were quite severe, with overall cost and minimized

program risks being primary considerations. No major aircraft modifications were allowed, the new rotor design had to be a “drop-in” replacement to ensure maximum compatibility with in-service engine components, and a weight increase of no more than 10 lbs. per engine was specified. These conditions were made all the more difficult by the limited availability of CIP funding from the USAF to accomplish this task. With increased CIP funding being highly unlikely, the USAF asked GE to consider a team approach to launch a viable design and production effort.

GE responded positively to this request and proposed a team consisting of the USAF, GE, and ABB Alstom to share the development cost. Due to the exceptional number of in-service eight-stage compressor engines around the world, there was a viable business case for a new rotor design outside of USAF purchases. This situation allowed the creation of a unique investment, certification, and data rights agreement that minimized the USAF’s CIP investment. Essentially, the agreement allowed the USAF to limit its investment to approximately 15% of the expected development cost upfront, essentially providing the seed money for the program. GE and ABB Alstom agreed to fund the remainder of the development and certification in exchange for exclusive rights to the eventual design. The design was to be certified to U.S. Federal Aviation Administration guidelines under FAR33 rules, with the proviso that GE and ABB Alstom provide additional information to the USAF under the CIP umbrella for analysis and verification. As a consequence, the new rotor would be made available in a commercial catalog, with standard commercial terms, conditions and pricing. The innovative approach sponsored by the USAF allowed the service to realize a huge return on a relatively small investment.

With the business side taken care of, attention turned to the task of designing a new rotor. Fundamentally, the airflow characteristics and aeromechanics of the current compressor design were exceptionally sound, and there was considerable reluctance to change the flowpath configuration in any new design. The decision was made to keep the current airfoil and flowpath contours, which would also allow use of the existing cases and stators with minimal risk. The primary need was to select a new rotor material that would eliminate the risk of corrosion, and marry it

to the spooled rotor concept developed for the -21 engine.

At first, it appeared that use of titanium as in the -21 would simultaneously address the corrosion issue as well as minimize any weight growth due to rotor design changes. However, there was no previous GE experience of using steel-based blades in a titanium spool for the military environment. This combination had been successfully fielded in commercial engines, but the increased cycling and operational stresses inherent in military use raised concerns over fretting taking place at the blade to rotor interface. While there was the potential to provide a protective coating at this interface, it was felt that this introduced an unnecessary element of risk and would have increased long-term costs in maintaining the coating. Replacement of the AM355 steel blades with titanium blades was also considered, but was rejected due to the larger axial clearances needed and the incompatibility of the existing inventory of blades with the new rotor. The best solution appeared to be the selection of a steel-based material with corrosion properties superior to AM355, which proved to be Inconel 718.

Inco 718 had previously been used as a vane material in the J85, and promised excellent resistance to corrosion. With a target of a 3:1 increase in the low cycle fatigue life limit using both probabilistic and deterministic methodologies, a design was produced that replaced the eight individual disks with one disk and two spools (Figure 6). This redesign resulted in a dramatic 8:1 reduction in parts count, a 5:1 reduction in projected maintenance man-hours, and eliminated high stress concentration rim bolt holes. It also provides the capability for individual blade replacement without rotor disassembly or rebalancing.

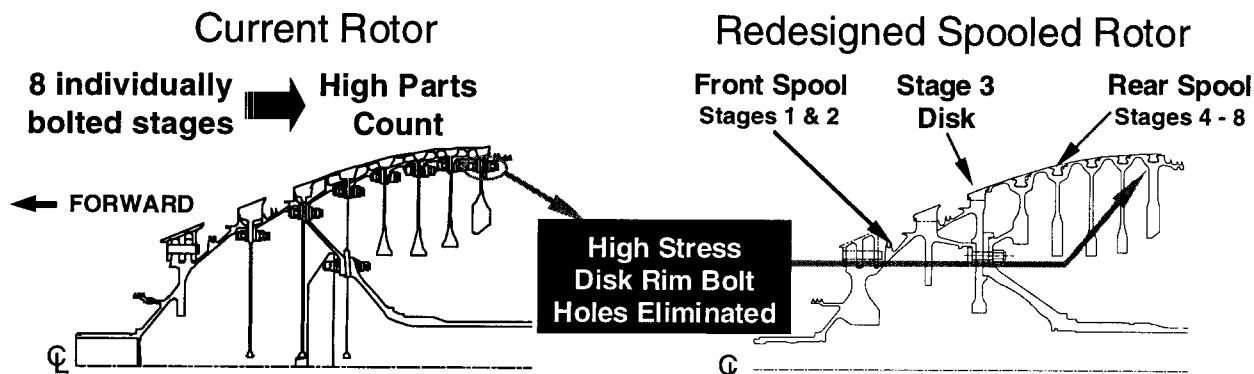


Figure 6: Design Comparison of Eight-Stage Compressor Rotors

However, by using Inco 718 versus titanium for the new spooled rotor design, the weight increased by 20 lbs. over the current rotor, exceeding the requirement by 10 lbs. The design was purposely conservative in order to be robust in service and have margin for the LCF life target, but could have been made somewhat lighter with further design effort. A study revealed the negligible impact of a 40 lb. propulsion system weight increase for the T-38, and the new design was accepted by waiver to the original and somewhat arbitrary weight requirement.

In keeping with the FAR33 certification process, the new compressor was not test flown on a military aircraft. Instead, the test vehicle was a Learjet 24/25 with one of its engines having been

modified with the new compressor. The civil CJ610-6 engine is closest in configuration to the J85-GE-5, and a -6 was modified and flown to provide the necessary data for certification. This testing was completed in early 2000, with a dirty and clean layout inspection of the hardware scheduled for later that year. In keeping with the risk mitigation process employed throughout the redesign effort, testing of the new compressor was accomplished both with new compressor cases and older production cases representative of those that would be encountered during the retrofit program. All of these test combinations have been successful, with a spooled rotor expected to be available as a standalone, drop-in replacement kit (Figure 7) in late 2000.

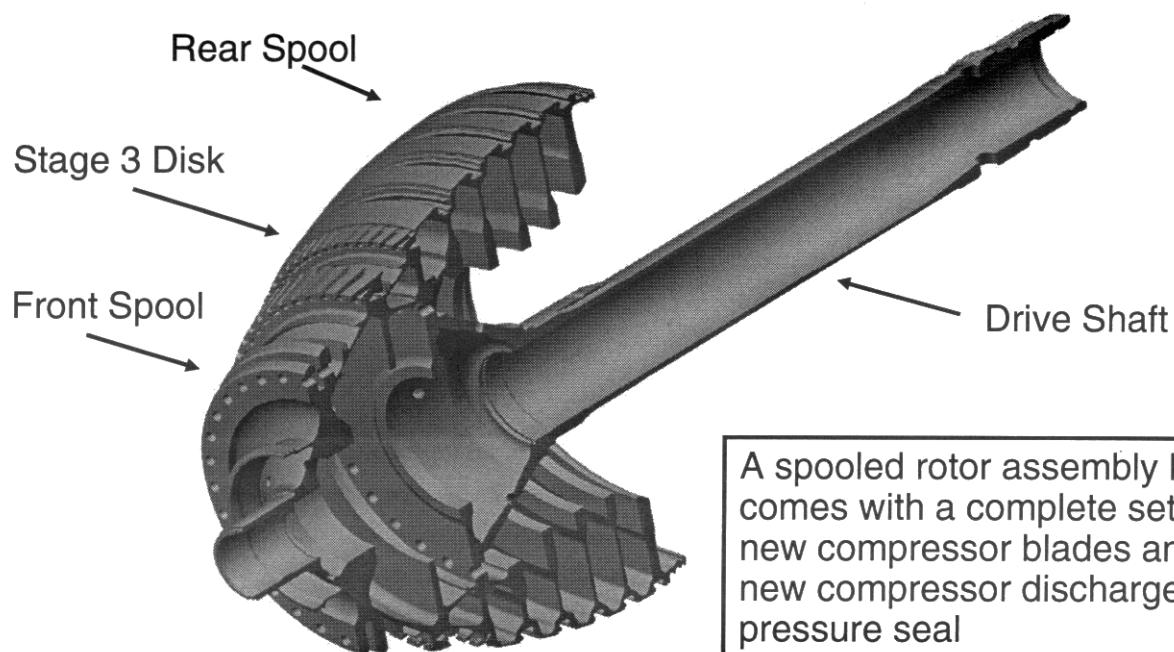


Figure 7: Redesigned Eight-Stage Compressor Rotor Assembly

Improvements Beyond the Compressor Rotor

The success of the initial look at updating the compressor rotor prompted the USAF to undertake a comprehensive review of what could be done to address other issues on the J85. This review centered developing a long-range propulsion roadmap that would be consistent with the T-38 structural and avionics upgrade programs allowing operations into the 2040 timeframe. The roadmap was to develop options that focused on reducing the cost of ownership, and increasing the overall safety, reliability, and durability of the propulsion system. These options fell into the broad categories of component improvements, engine module upgrades, aircraft system enhancements, maintenance-related changes, and even the possibility of re-engining. Table 2 lists some of the items analyzed during the propulsion roadmap study.

As with the spooled rotor redesign effort, funding to develop and acquire these upgrades was in short supply. Each task was analyzed for return on investment and life cycle cost benefit, and then balanced against the available funding. The net result of this study was the definition of a set of upgrades for the T-38C that would collectively be known as the J85 Propulsion Modernization Program, or PMP. At the heart of this upgrade program was the new eight-stage spooled compressor rotor described previously. Figure 8 identifies the major kits that comprise the PMP. While some of the upgrade items in Table 2 were not selected for the PMP, they remain viable candidate programs to address other customers' requirements for J85 improvements, while several more programs exist for the -21 version of the J85.

Components	Modules
Long Life Combustor	Spoiled Compressor Rotor
Cast Nozzle Flaps & Seals	Mini-Growth Turbine
Stacked-Ring Afterburner Liner	New Design Turbine
Ignition Plugs & Leads	New Turbomachinery
Variable Geometry Actuator	New Design Compressor
Compressor Case	
Ignition Exciter	
Afterburner Fuel Pump Upgrade	
Main Fuel Pump Upgrade	
Hydraulic Nozzle Actuation	
Afterburner Fuel Control	
Digital T5 Amplifier	
Mainframe Upgrade	
Full Authority Digital Engine Control	
T2 Sensor Replacement	
Compressor Bleed Valves	
	Aircraft Systems
	Ejector Nozzle
	Inlet Lip Modification
	Maintenance
	T5 Motor Overhaul
	Overspeed Governor Overhaul
	Master Chip Detector
	Heatshield Access Port to Igniter
	Re-engining
	New Centerline Engine
	Alternate Engine

Table 2: T-38/J85 Propulsion Roadmap Items

The PMP wasn't just a program to put old part designs back into production, making new "old" production hardware. The opportunity to analyze significant amounts of field data and apply state of the art materials, designs, and production techniques was taken with a vengeance.

In conjunction with providing a new compressor rotor, new stator cases were made available with increased corrosion resistance. This was accomplished by changing from AM355 base material with coating in the stator lands to Inconel as the base material, eliminating the need for coatings and their high-maintenance care.

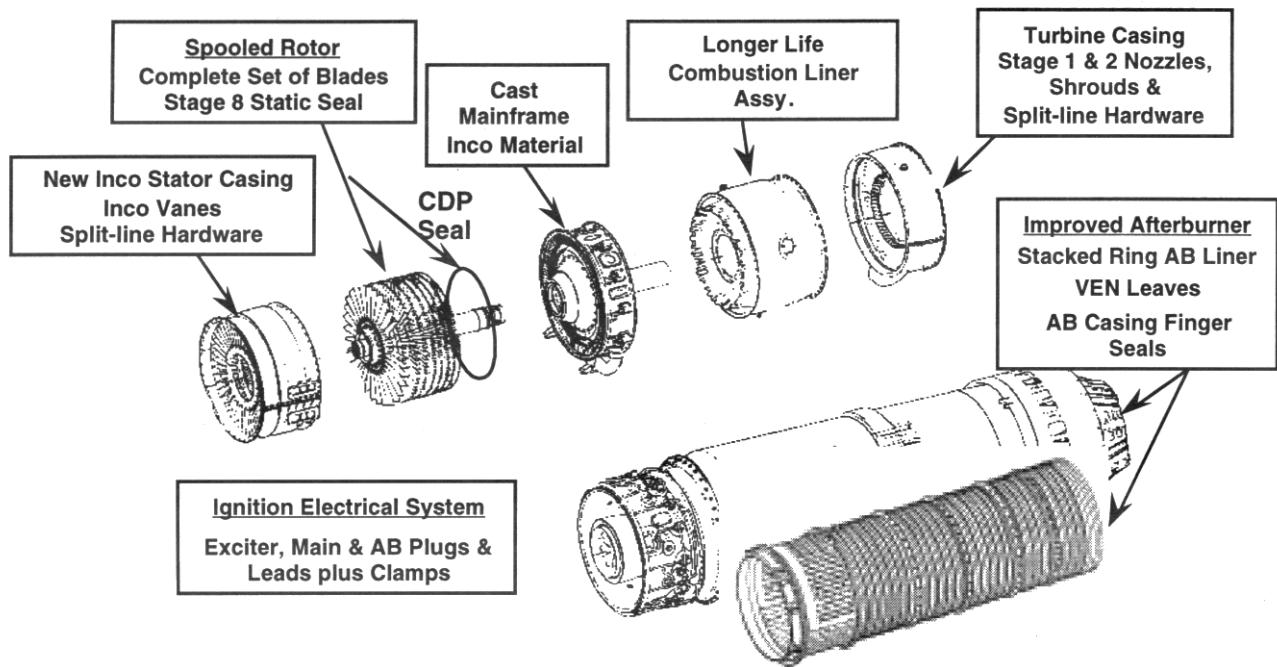


Figure 8: J85 Propulsion Modernization Program Upgrades

A new single-piece cast mainframe was created to replace the original 39 piece welded design, as well as introducing Inconel for improved corrosion resistance. The welded design was susceptible to fatigue cracking that originated along the edges of the welds joining the struts and pads to the outer shell. While repair of the cracks is possible, it is expensive both in cost and engine down time.

Taking advantage of new materials and coatings allowed a significant increase in the reliability of the J85 combustor. By applying revised thermal barrier coatings (TBC) and placing it in additional areas of the combustor, both the service life and inspection intervals have been increased by a factor of 2 to 1.

Addressing the traditional high distress area of engines was accomplished by providing a comprehensive upgrade kit for the high pressure turbine section. A new turbine casing with revised Stage 1 and 2 nozzles was able to reduce cracking by porting cooling air to new areas. New shroud and turbine blade materials increased the temperature capability, allowing greater time on wing.

Decreasing the maintenance costs and increasing the reliability of the afterburner was a

major focus of the PMP, as liner cracking was the leading cause of unscheduled engine removals. The current liner design consists of 4 major sections and containing over 900 parts in total. The brackets used to support the liner produced high stress concentrations. A new stacked ring liner was designed that is a single major piece, with less than 100 additional parts required to complete the assembly. The bracket system was eliminated entirely by transitioning to a free-floating design, and TBC was applied in the screech section. Durability for the liner alone was increased by a factor of 6 to 1. New variable exhaust nozzle (VEN) leaves and afterburner casing seals, changed as a set, have revised chrome carbide and electroless nickel coatings for a 3X service life improvement.

Perhaps more mundane but no less critical, the issue of electrical parts obsolescence was tackled in the engine ignition system. Certain ignition exciter components are no longer available at any price, and the overall design was seriously outdated (1950's electronics technology). By utilizing a modern exciter with new plugs and leads, overall ignition energy and system reliability is dramatically increased, resulting in a reduction of no-starts and afterburner no-lights by over 50%.

System Level Improvements

The J85 engine is but one component of the T-38 aircraft propulsion system, and the entire system was evaluated in developing technology options for improved performance and reduced cost of ownership. Two items identified in the propulsion roadmap fall into this category of improvement, the ejector nozzle and the inlet lip modification.

The original configuration of the T-38 nozzle ejector was sized for the maximum afterburner condition. This exit area caused over-expansion of the exhaust plume at Military power, with the result being excess drag and lower overall

net thrust. A new system was designed that incorporated free floating doors in an ejector that was a simple replacement of the current hardware. The aerodynamic actuation system resulted in an effective variable size at all power conditions, more efficiently controlling the plume expansion. This provides a system level benefit in terms of higher net thrust of 1-2% at takeoff conditions and up to 10% at Military power, resulting in decreased fuel burn or increased aircraft performance. In addition, the stall-free engine envelope was increased, and the nozzle components were exposed to a decreased temperature environment. Figure 9 shows the basic configuration of the revised ejector design, and Figure 10 compares the installation of the revised design to the current.

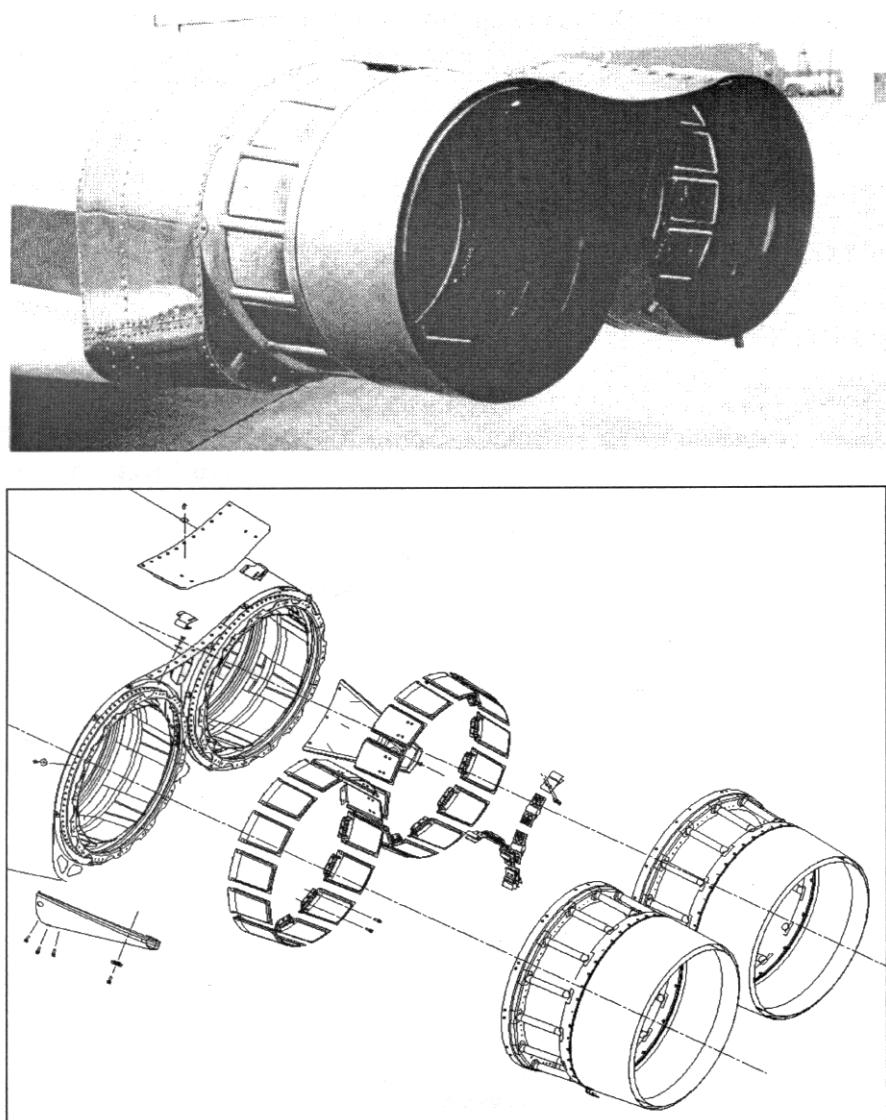


Figure 9: Redesigned Ejector Hardware

The redesigned ejector nozzle is shown installed on a NASA T-38, which accomplished a series of risk-reduction flight tests to validate the projected performance characteristics and improvements. The results of the flight test are shown in Figure 11 plotted on an aircraft flight map with areas of anticipated performance increases displayed, confirming the analytical model of the ejector.

This significant performance increase could be even further enhanced with the incorporation of a revised inlet lip contour. The T-38 was the first

aircraft to go into production in Northrop N-156 family, and subsequently aircraft models used modified inlet configurations to improve engine performance and stall margin. Using this design history as a base, a new inlet shape incorporating a fatter lip with a squarer profile was developed. This new design was also tested on a NASA T-38 in the form of an add-on glove to simulate replacement of the current design, resulting in a startling 22% thrust increase at take-off conditions, with improved stall margin throughout the envelope.

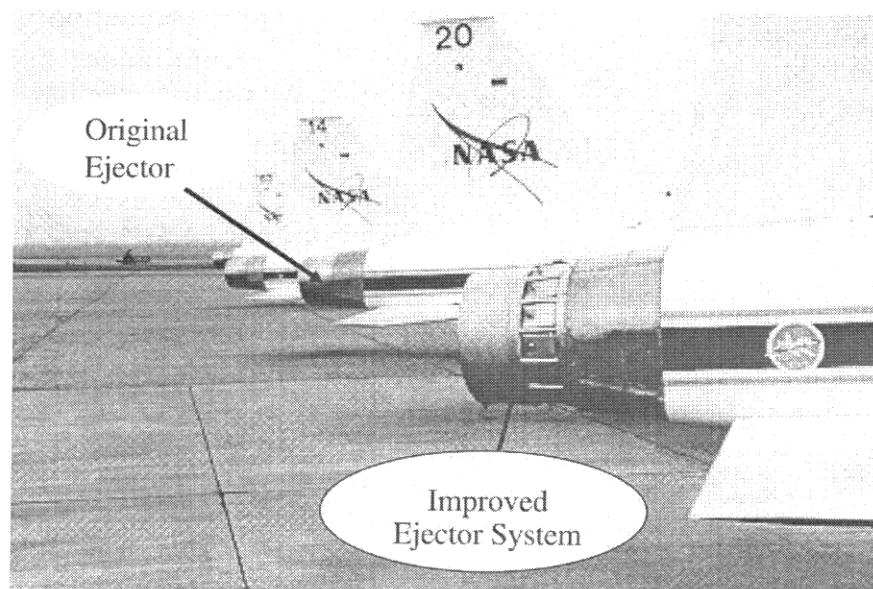


Figure 10: Comparison of Original Ejector to Improved Ejector System

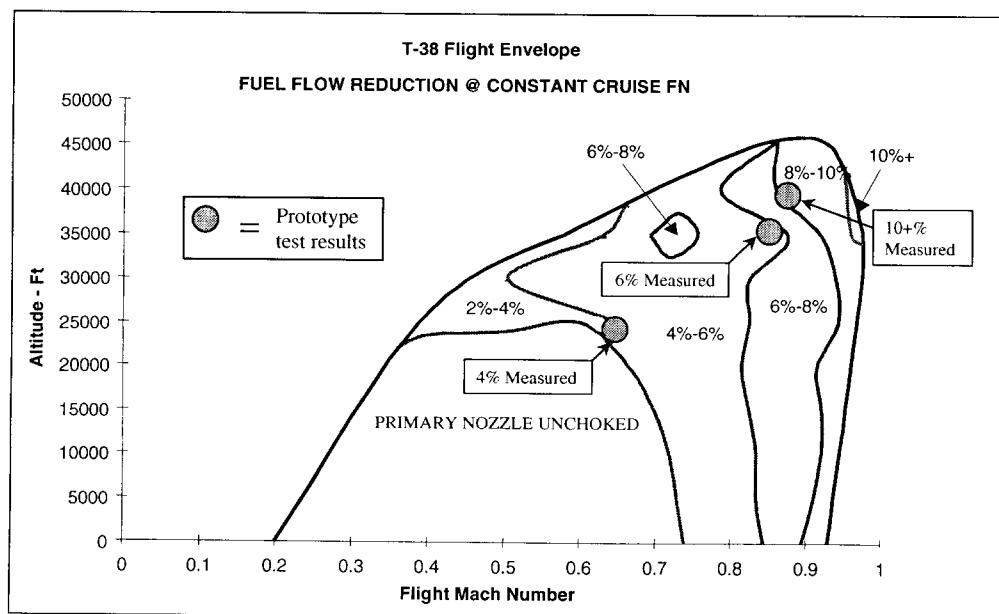


Figure 11: Ejector Nozzle Flight Test Results

Summary

The technologies and design capabilities available for new engines today were only a gleam in propulsion engineers' eyes 50 years ago. The performance, operating costs, inspection intervals, and service life of engines designed in that timeframe pale in comparison to modern military engines. But those modern tools, combined with a rigorous and honest analysis of fielded engine experience, can produce technology upgrade options that rejuvenate engine performance and reliability, yet are affordable to acquire.

The J85 engine is a classic example of an engine with very modest beginnings being called upon to serve longer and used much more differently than was ever originally imagined. With acquisition of new aircraft all but impossible due to continually decreasing military budgets, soldiering on with existing equipment is the order of the day. A committed manufacturer, seeing with the eyes his customer, cannot help but to support the continued operation of aging engines with the technological and intellectual resources at their disposal. Indeed, GE Aircraft Engines has embodied this spirit of commitment in its internal policies, stating "GE Aircraft Engines (GEAE) will continue to provide support for all GEAE products for as long as any operator continues to use the products." By rationally packaging kits of engine upgrades that increase time on wing, scheduled inspection intervals, and overall system performance, it becomes possible for users to break the death spiral of increasing maintenance costs.

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THE AGING OF ENGINES: AN OPERATOR'S PERSPECTIVE

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ABSTRACT

NATO countries are currently faced with the need to operate fleets of mature gas turbine engines built many years ago. Because of diminishing resources for new equipment, the prospects of replacing these engines with new ones are not good at present. How long such engines can be kept in service safely, without replacing a significant portion of their aging structural components has become a growing concern to engine life-cycle managers, due to uncertainties in residual lives. Another concern is the high maintenance cost associated with the replacement of durability-critical components, such as blades and vanes. The need to balance risk and escalating maintenance costs explains the growing interest in the application of life extension technologies for safely extracting maximum usage out of life-limited parts. In the case of aero-engines, maintaining airworthiness while ensuring affordability is of prime concern to both life-cycle managers and regulatory authorities. This lecture describes the modes of deterioration of engine components and discusses their effects on the performance, operating costs, reliability and operational safety of engines. It also identifies component life extension strategies that engine life-cycle managers may adopt to cost-effectively manage their engines, while ensuring reliability and safety. A qualification methodology for component life extension, developed and implemented for Canadian Forces engines, is presented. The methodology incorporates an Engine Repair Structural Integrity Program (ERSIP) that was conceived to establish structural performance requirements and identify tests for development and qualification of life extension technologies, to ensure structural integrity and performance throughout the extended life. Examples of life extension technologies applied to gas path components and critical rotating parts are described, including the use of protective coatings and repairs to increase component durability. The application of damage tolerance concepts to allow use of safety-critical parts beyond their conventional safe-life limits is also illustrated.

GLOSSARY OF ABBREVIATIONS

AET	Accelerated Endurance Tests
AMT	Accelerated Mission Test
ASMET	Accelerated Simulated Mission Endurance Test

CG	Columnar grain
CIP	Component improvement program
DFM	Deterministic fracture mechanics
DS	Directionally solidified
EBW	Electron beam welding
ENSIP	Engine Structural Integrity Program
ERSIP	Engine Repair Structural Integrity program
FCGR	Fatigue crack growth rate
FEM	Finite element modeling
FM	Fracture mechanics
FMECA	Failure Modes and Effects Criticality Analysis
FOD	Foreign object damage
HCF	High cycle fatigue
LEFM	Linear-elastic fracture mechanics
LCF	Low cycle fatigue
LCM	Life cycle management
LCMM	Life cycle materiel manager
LPI	Liquid penetrant inspection
NDI	Non-destructive inspection
OEM	Original equipment manufacturer
PFM	Probabilistic fracture mechanics
POD	Probability of detection
R&O	Repair and overhaul
SC	Single crystal
SIF	Stress intensity factor
SII	Safe inspection interval
TBC	Thermal barrier coating
TCP	Topologically closed packed
TF	Thermal fatigue
TIT	Turbine inlet temperature

1. INTRODUCTION

Turbine engine components accumulate damage in service as a result of their demanding operating conditions. The damage may take many forms depending on the component, engine type and operating environment [1]. Service-induced damage limits the usable life of many engine components. When damage becomes excessive as revealed by inspection or when component design lives are reached, the components are replaced with new ones. Replacing service-damaged parts is costly and is a significant contributing factor in the overall life cycle cost of engines.

Annual expenditures on replacement parts for modern engines can be well in excess of \$50,000 per engine [2].

Occasionally, that cost may grow significantly when components such as compressor wheels or turbine discs need replacing because the specified safe-life for the parts has been reached. Thus, at a time of tight budgets and diminishing resources for new equipment acquisition or equipment maintenance, it is not surprising that many NATO engine fleet managers have been exploring every possible way to extend the usable lives of expensive components. In the case of old engines, in particular those that were designed more than 40 years ago, the fleet managers may have to face additional challenges due to the scarcity of spare parts or spares not being available. In the absence of spare parts, uncertainties in component residual lives may be cause for concern.

For engine life cycle managers to ask the right questions and make the right decisions, they need to clearly understand how engine components deteriorate in service and how this deterioration impacts on the performance, operating cost, reliability and safety of engines. They also need to know how to take advantage of the latest developments in component life extension technologies to cost-effectively manage their engines. In this respect, it is important for fleet managers to be fully cognizant of the requirements to qualify the life extension technologies for ensuring the engines remain reliable and safe to operate upon return to service.

The objective of this paper is to provide engine managers with guidance and strategies for the development and implementation of new or existing life extension technologies. It describes the modes of deterioration of engine components and how the damage affects structural integrity. It reviews procedures applied to manage aging components and some of the latest developments in component life extension. It explains the Qualification Methodology developed jointly by the NRC Institute for Aerospace Research (IAR) and Orenda Aerospace Corporation (OAC) for the Canadian Department of National Defence (DND). Finally, it provides examples of substantiation test programs that have been used to qualify some recently developed life extension schemes for CF engines.

2. MODES OF DETERIORATION OF ENGINE COMPONENTS

Damage incurred by engine components during service may be external, affecting dimensions and surface finish as a result of fretting, wear, erosion, corrosion or oxidation. It may also be internal, affecting the microstructure of highly stressed and hot parts, as a result of aging of the material microstructure, creep or fatigue. External damage impacts significantly on functionality of the parts, including the aerodynamic performance of gas path components, but may also reduce their load bearing capacity. Surface damage in the form of low cycle fatigue (LCF) cracks, scars or dents, fretting-wear or foreign object damage (FOD), respectively, may lead to high cycle fatigue (HCF) failures [3]. Internal damage may reduce component strength and lead to component distortion. Its accumulation also causes the initiation of flaws, which may ultimately lead to cracking and component failure. Examples of external and internal

forms of damage and component distortion and cracking are presented below.

2.1 Surface Damage

2.1.1 Fretting is caused by oscillatory motion of very small amplitude between two contacting surfaces. It occurs in engines, for instance, where blades come in contact with discs. The rubbing motion between the components generates debris that is trapped between the two contacting surfaces and may lead to scoring. The resulting surface scar may act as a stress raiser and give rise to fretting-fatigue cracks. Both fan and compressor sections are susceptible to this form of damage. The susceptibility to fretting of blades and discs is governed by many geometric and materials factors. The latter include the difference in hardness between the blade and the disc materials, the magnitude of surface residual stresses and the lubrication of the contact surfaces. Depending on the hardness ratio either one of the components may incur the damage. A close up view of fretting damage incurred by the dovetail of a compressor blade is shown in Fig. 1.

Fretting tends to reduce HCF life, but may also impact on LCF life of the parts. Fretting may also affect splines, couplings, clutches, spindles and seals. It may occur in joints that are bolted, keyed, press fitted, shrunk or riveted. Fretting can be minimized through use of anti-fretting compounds, including soft coatings, and surface modification treatments [4].

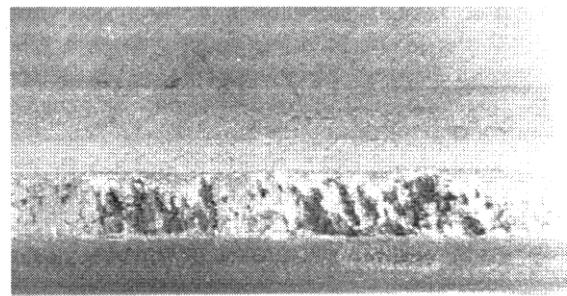


Figure 1. Dovetail of a compressor blade from a CF engine subject to fretting damage, showing deep fretting scar. The surface damage may lead to HCF or LCF failures.

2.1.2 Erosion of compressor airfoils is caused by ingestion of hard particulate matter through the engine air intake. The impacting action of gas entrained hard particles on gas path components leads to gradual removal of material, which given sufficient service time can significantly alter component shape and surface finish. The change in geometry can be quite severe as evidenced by the eroded airfoils of the compressor vane segment shown in Fig. 2. Change in leading edge radius and surface roughening can significantly reduce compressor efficiency and engine performance. In extreme cases, geometry changes may alter the modal response of the airfoils, to the point where they may be excited at their natural frequencies under normal operating conditions. Such resonant vibration may lead to HCF failures of blades. Erosion of compressor gas path

components can be minimized through the use of inlet particle separators or by enhancing the erosion resistance of airfoils through use of protective coatings or surface modification treatments [5].

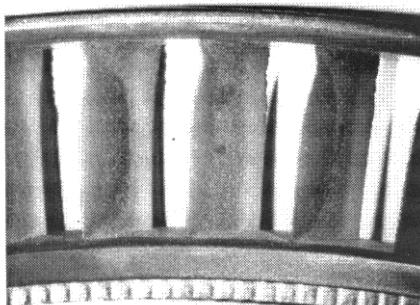


Figure 2. Erosion damaged CF T56 compressor vane segment after approximately 5000 hrs of service.

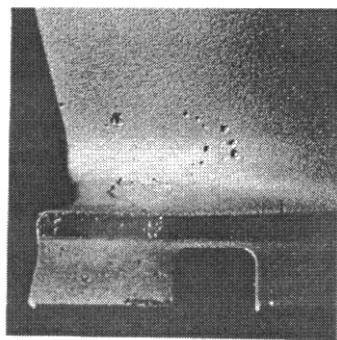


Figure 3. Corrosion pits close to the root of a compressor blade exceeding damage allowable limits.

2.1.3 Pitting corrosion may occur in the presence of environments containing a high concentration of chloride ions, for instance in a marine environment. Both stainless steel and titanium alloys are particularly susceptible to this form of damage. Corrosion pits can have a dramatic effect on the structural integrity of compressor parts. Depending on the type of component and the location of the corrosion pits, they may act as crack initiation sites for HCF or LCF cracks. Corrosion pits of the type shown near the root of the compressor blade, Fig. 3, would normally be cause for blade rejection [6]. If left unchecked, this form of damage may lead to catastrophic component failure. Corrosion control may be achieved through use of protective coatings[7].

2.1.4 Oxidation damage is cause for rejection of a significant fraction of turbine blades and vanes during scheduled engine maintenance. Oxidation reduces the thickness of airfoils and their load bearing capacity. It may also give rise to sites for HCF or LCF crack initiation. In cooled blades, oxidation of normally uncoated internal cooling passages may combine with external surface oxidation to reduce component wall thickness and, thus, the load bearing capacity of the blade [8]. The coatings that are applied to most of these parts, over their airfoil portions, provide some protection against high temperature oxidation. The coatings are

aluminide intermetallic compounds designed to form a protective oxide scale that prevents ingress by oxygen and further oxidation. However, coating life is limited due to spalling of the protective oxide scales and gradual consumption of the coating elements. Thus, ultimately, even the coated parts need replacing.

Components made of stronger superalloys are often subjected to higher operating temperatures than their weaker counterparts and, therefore, they are exposed to more aggressive oxidizing environments during service. Thus component made of directionally solidified (DS) alloys with either columnar grained (CG) or single crystal (SC) microstructures are often life limited due to surface oxidation. The problem can be quite severe at turbine blade tips, because any protective coating wears off rapidly at this location, due to tip rub. In the case of CG blades, tip oxidation may lead to thermal fatigue cracking of the longitudinal grain boundaries that are embrittled as a result of boundary oxidation, Fig. 4 [9].

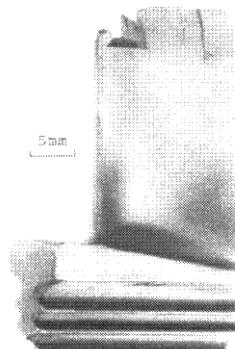


Figure 4. Tip cracking of a DS-CG blade caused by thermal fatigue of longitudinal grain boundaries embrittled as a result of grain boundary oxidation.

Blades and vanes may be recoated in conjunction with other refurbishment or repairs performed during engine overhaul. Tip repairs are an option for CG and SC blades [10].

2.1.5 Hot corrosion is a form of accelerated high temperature oxidation/sulphidation that may destroy hot gas path components when they are exposed to deleterious mixtures of reactive contaminants originating from the fuel and the ingested air. At temperatures in the range from 850 to 950°C, residual sulfur from the fuel combines with sodium chloride picked up from a marine environment to form sodium sulfate. This corrosive compound fluxes protective oxide scales, which destroys coatings, thereby exposing the substrate material to the environment. Once a hot corrosion reaction starts, it will quickly consume the component as shown in Fig. 5 [8]. Substituting a protective coating for another that has more resistance to hot corrosion may be a cost-effective option. The addition of a coating to the internal cooling passages of blades and vanes is also an option when hot corrosion occurs within the cooling passage [11].

2.2 Internal Damage

Another form of deterioration of engine components results from the accumulation of internal microstructural damage. This is an insidious form of damage because, in contrast to surface damage, it cannot be readily detected by non-destructive inspection (NDI) techniques. Its rate

of accumulation is strongly influenced by service stresses and temperature and, therefore, to a great extent by user practice. Because there are uncertainties in the temporal variation of service stresses and temperatures, internal microstructural damage cannot be easily predicted.

Internal microstructural damage is the result of plastic strain accumulation and metallurgical aging reactions [1]. Plastic strain accumulation is the product of fatigue and/or creep, which affects highly stressed and hot parts, such as compressor and turbine discs, as well as turbine blades and vanes. Metallurgical aging reactions affect mostly hot gas path components. They may also occur in the rim of discs in small turbines, which often operate at higher operational speeds and temperatures than do larger turbines [9].

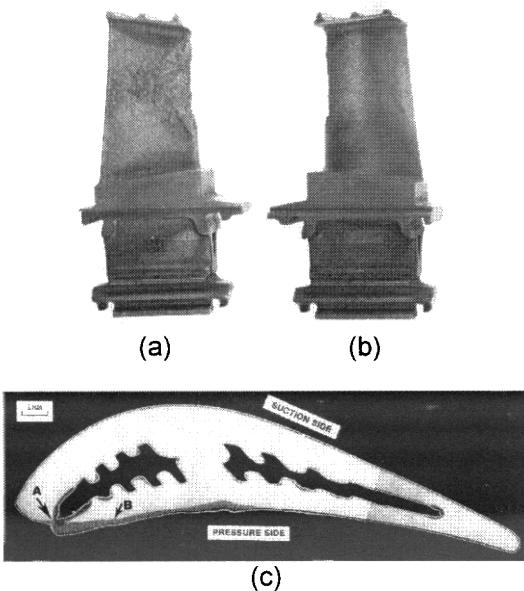


Figure 5. Hot corrosion damage incurred by a Mar-M246 blade in the engine of a CF maritime patrol aircraft (a) pressure side view; (b) suction side view; (c) metallographic section taken halfway across the airfoil through the hot corroded region [8].

2.2.1. Plastic strain accumulation manifests itself in the form of dislocation substructures, including persistent slip bands and wavy slip, which develop under cyclic thermo-mechanical loads at stress concentration sites such as disc bores and bolt holes or the rim region of rotors, prior to crack initiation. The dislocation substructure shown in Fig. 6(b) is indicative of high temperature LCF damage accumulation in turbine discs [12]. Such form of micro-damage accumulation is mostly dependent on number of applied load cycles. Plastic strain accumulation also manifests itself as creep deformation leading to formation of creep cavities and internal cracks, Fig. 6(d) [1].

2.2.2 Metallurgical aging reactions include coarsening, agglomeration and rafting of gamma prime precipitates in nickel base superalloys, carbide coarsening in cobalt base superalloys, degeneration of primary carbides into continuous carbide films along the grain boundaries of

polycrystalline alloys, and precipitation of topologically closed packed (TCP) phases, such as sigma and/or mu phases within the interdendritic regions of the alloys [1]. Such reactions are mostly time dependent, although to some degree some can be stress assisted as well. For instance, in DS CG or SG alloys, the stress field within components dictates the orientation of gamma prime precipitate rafting.

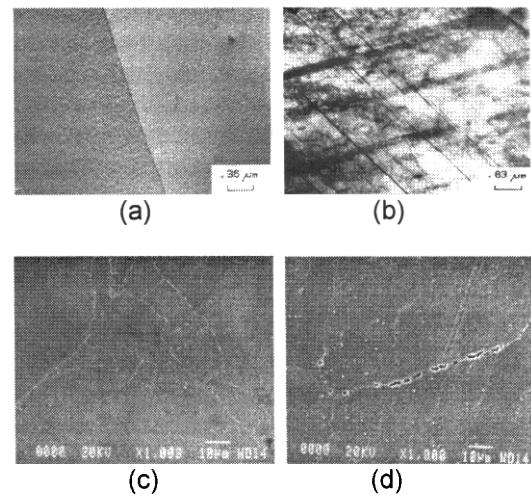


Figure 6. Effects of service exposure on microstructure (a) Virgin disc; (b) Service exposed disc showing evidence of dislocation activity (PSB) indicative of LCF damage accumulation (c) Virgin disc spacer; (d) Service exposed disc spacer showing evidence of creep voids along grain boundaries; [1,12].

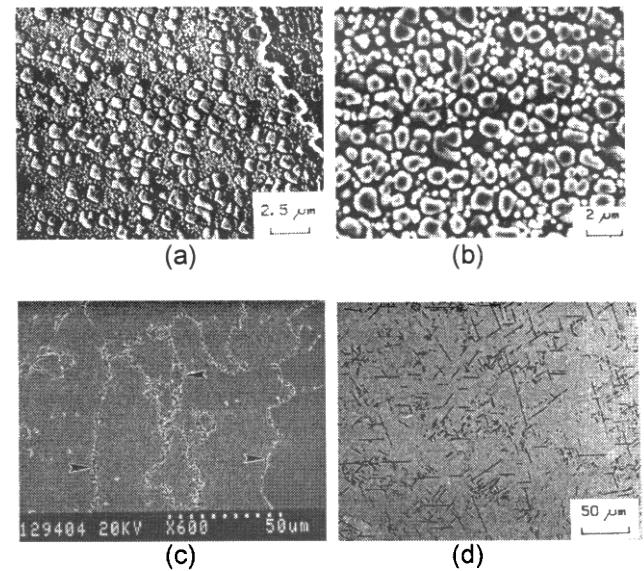


Figure 7. Internal microstructural damage in high time blades: (a) bimodal distribution of gamma prime precipitates in new blade; (b) coarsening of gamma prime precipitates and elimination of secondary gamma prime in service exposed IN738 turbine blade; (c) precipitation of carbides (arrows) along grain boundaries in IN738 turbine blade; (d) precipitation of sigma phase in IN 713C blade [13,14].

Examples of gamma prime coarsening, carbine reactions and TCP phase precipitation in nickel base superalloys are shown in Fig. 7 [13,14]. Precipitate coarsening and TCP phase formation reduce material strength, whereas continuous carbide film formation and TCP precipitates embrittle the material, therefore, making components notch sensitive. The precipitation of sigma phase immediately below the protective coatings, Fig. 8, compounds the notch sensitivity problem during service.

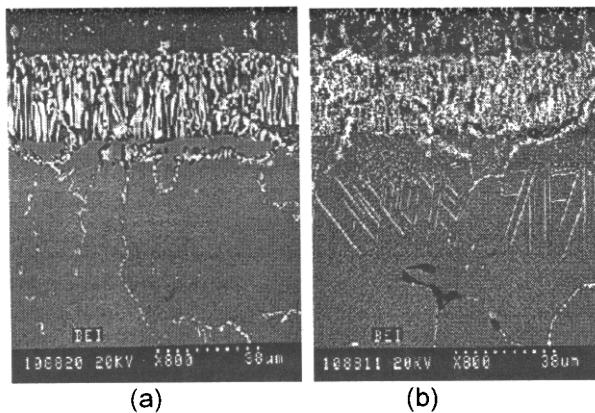


Figure 8. Service-induced precipitation of sigma phase immediately below the protective coating in a Mar-M246 turbine blade from a CF transport aircraft engine; (a) new blade; (b) service-exposed blade.

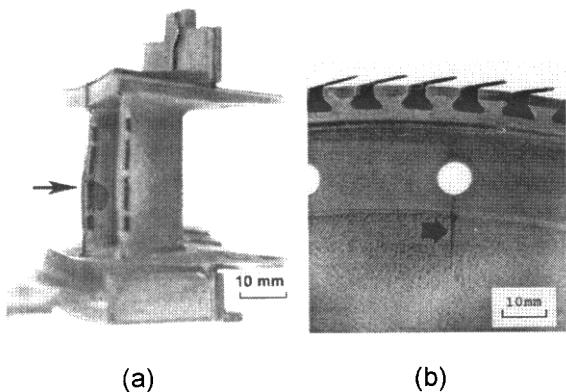


Figure 9. (a) Airfoil bowing, loss of coating and cracking of a CF T56-A15 first stage nozzle guide vane; (b) Crack initiating from bolt hole in CF j85 CAN40/15 compressor disc [15, 16].

2.3 Distortion and Cracking

Under creep loading conditions, the loss of strength caused by service-induced metallurgical ageing leads to component distortion. Vane airfoil bowing or lengthening and untwist of turbine blade airfoils is commonly observed in aero engines Fig. 9(a) [15]. Extreme airfoil distortion may lead to HCF failures. Cracking due to cavity link-up, or oxidation-assisted creep cracking along the grain boundaries, usually follows microstructural damage accumulation, in conjunction with component distortion. Microstructural damage accumulation under LCF loading conditions is also followed by crack initiation and growth in highly stressed components such

as discs, Fig 9(b) [16]. This type of cracking may lead to catastrophic failures.

3. MANAGING THE DETERIORATION OF ENGINE COMPONENTS

For the purpose of life cycle management, engine components may be classified as either durability-critical or safety-critical [17].

(a) *Durability-critical components*, includes those whose deterioration affects mainly engine performance and fuel efficiency and may result in a significant maintenance burden but will not normally impair flight safety. These parts include cold and hot gas path components such as blades and vanes. The life of these parts is limited by one of a variety of possible modes of damage accumulation, which vary with the type of component and its operating environment.

(b) *Safety-Critical components*, includes those whose fracture may result in the loss of the aircraft if the fracture is not contained. These components include most of the large rotating compressor and turbine components, such as wheels, discs, spacers and shafts. The life of these parts is usually limited due to LCF damage accumulation.

For *durability-critical parts*, life limits are rarely specified. An “*On-Condition*” maintenance approach is normally used, where parts are removed from service when measurable damage limits are reached [18]. Damage limits are specified in R&O Manuals, including for instance limits on the extent of corrosion pitting, elongation or untwist of turbine blades or cracking of turbine vanes. No “hard time” life is set for these parts, although a minimum life expectancy may be guaranteed by the original equipment manufacturer (OEM) for turbine blades and vanes. However, the difficulties associated with predicting service behaviour of metallurgically complex blade and vane materials, under conditions that can vary widely with user practice, make it difficult to reliably establish life expectancies. These difficulties are made worse by complex coating-substrate interactions that are not usually well understood. Consequently, it is not unusual for shortfalls to be experienced in life expectancies of turbine blades or vanes.

For *Safety-Critical parts*, whose lives are limited by LCF, two life cycle management approaches may be used [18]:

- (a) The “*Safe-Life*” approach, for which all components are retired before a first crack is detectable, and
- (b) The “*Damage Tolerance*” approach, for which component lives are established on the basis of fracture mechanics (FM) principles and retirement of parts is based on a number of life cycle management (LCM) options, as explained below.

3.1 The Safe Life Approach

The Safe-Life approach for LCM of safety-critical parts assumes that when a crack appears, the component has

failed, and it further ensures that all similar components are retired at an equivalent life, before the first crack appears. To that end, the approach follows a “cycles to crack initiation” criterion, with a minimum safe life (also called hard time life) capability established statistically through extensive mechanical testing of test coupons and components under simulated service conditions. The statistical minimum is based on the probability that only 1 in 1000 component (-3σ) will have developed a detectable crack, typically 0.8 mm in surface length, at retirement, Fig. 10. This approach is known in the UK as the life-to-first-crack (LTFC) method, for which the detectable crack radius is chosen to be 0.38mm.

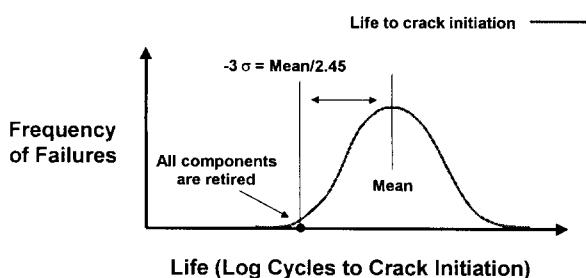


Figure 10. The Safe-Life approach.

The advantages of the Safe Life approach are that the maintenance requirements are kept to a minimum, while the time in service of components without inspection is maximized. The main disadvantage of the Safe-Life approach is that it may be overly conservative, because components are retired with a significant amount of useful residual life (by default, 999 out of 1000 components are retired with significant life remaining) [17,18]. Furthermore, the method is costly since all parts may need replacing nominally all at the same time. Finally, the availability of spares may be a problem for old engines.

3.2 The Damage Tolerance Approach

The Damage Tolerance approach assumes that fracture critical areas of components contain crack-like manufacturing or service-induced defects giving rise to the propagation of cracks during service. It further assumes that components are capable of continued safe operation as the cracks grow under thermal and mechanical stresses. Cracks are assumed to grow in a manner that can be predicted from linear-elastic fracture mechanics (LEFM), or other acceptable methods. Cracks are also assumed to grow sufficiently slowly to allow their detection through regularly scheduled inspections. Finally, the approach follows an inspection schedule established by analysis to ensure that cracks will not grow beyond a dysfunction limit, in between inspections.

The time interval between the scheduled inspections, or *safe inspection interval (SII)*, is calculated as the time it takes a hypothetical crack to grow from a size immediately below the detection limit, to a dysfunction crack size, beyond which the risk of rapid or unstable crack growth becomes too high. The dysfunction crack size is obtained for an assumed crack geometry from the

fracture toughness of the material and the crack tip stress intensity factor, using appropriate safety factors. In practice, the SII is assumed to be one half the safety limit. The approach is described schematically in Fig. 11.

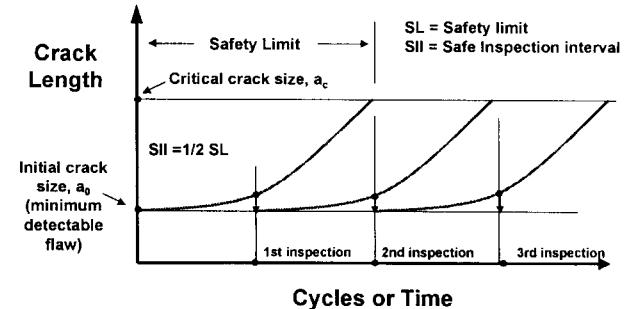


Figure 11. The Damage Tolerance approach.

For damage tolerance based LCM of safety-critical parts, two implementation methods may be used, one covered under MIL-STD 1783 (ENSIP) and the other known as Retirement for Cause (RFC) [18]. A third option known as 2/3 Dysfunction Life may also be practiced [19]. Each approach is explained below.

3.2.1. ENSIP (MIL-STD-1783)

The Engine Structural Integrity Program (ENSIP) is an organized and disciplined methodology introduced in 1984 by the USAF for the structural design, analysis, development, production and life management of engines. ENSIP embraces a damage tolerance approach to set *safe inspection intervals* for life management of safety-critical parts. However, conventional structural design criteria are also used to minimize risks of failure due to vibration, LCF, HCF and creep. Individual parts (e.g. compressor and turbine discs) are retired once their safe-life (LCF life to crack initiation) is reached, Fig 12(a) [18].

3.2.2 Retirement for Cause

The Retirement for Cause concept is a damage tolerance based method for managing the life of safety critical parts. The method relies on fracture mechanics to set a Safe Inspection Interval (SII) as practiced under ENSIP. Retirement life is based on periodic inspections until a crack is detected, at which point the part is retired. Individual parts, (e.g. compressor and turbine discs) are retired once they are found to contain a crack, Fig 12(b) [18].

The advantages of a damage tolerance based approach for LCM of safety critical parts are twofold. Firstly, the approach ensures that cracks emanating from manufacturing defects (or service induced cracks) in any one of the components will not grow beyond dysfunction size. Secondly, it allows life extension beyond LCF safe-life limits if and when needed, through use of the RFC technique [17,18].

However, there are also some disadvantages. The damage tolerance approach is more costly to implement than the safe life approach. It requires an elaborate NDI infrastructure to support increased inspection

requirements. Also, the handling of components is increased. Finally, in the context of new engine designs it may add weight to components that are conceived for damage tolerance. Such components may require thicker sections to better tolerate the presence of cracks or to ensure low crack growth rates [17,18].

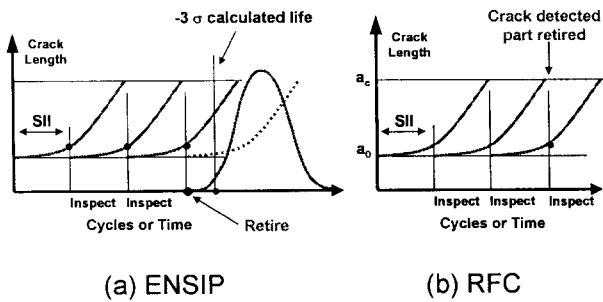


Figure 12. Schematic representations of the damage tolerance approaches based on (a) MIL-STD 1783 (ENSIP); (b) Retirement for Cause (RFC).

3.2.3 The 2/3 Dysfunction Life Approach

With this approach, developed in the UK, an acceptable but variable margin of safety is given by replacing the detectable “engineering crack” of the Safe-Life approach by a set fraction of “2/3 of the total life to dysfunction”, that is the life before the onset of rapid crack growth. The value of 2/3 is chosen because experience has shown that the crack size at “2/3 failure” is approximately equal to 0.38 mm, as defined in the LTFC method. For damage tolerant designs, in which the 2/3-dysfunction life exceeds the LTFC, this approach provides opportunity for life extension. For such cases, FM crack propagation calculations may be used to determine the available service life beyond “first crack” [19].

Whatever is selected as the LCM approach, safe and reliable prediction of component cyclic lives or safe inspection intervals requires an accurate knowledge of the stresses and temperature in the parts, as well as their variation with time. A detailed knowledge of the response of structural materials to external forces and the environment, especially in terms of active modes of damage accumulation and their interactions is also important. Experience shows that there are significant uncertainties associated with all these external variables, as well as with the accuracy of structural analysis methods and damage modeling methods employed for predicting lives.

4. MANAGING NATO'S AGING ENGINE FLEETS

Several of the NATO countries are faced with having to operate old engines for which the extent of damage incurred by fracture-critical parts, and therefore the residual lives of the parts, are not accurately known. Owing to diminishing resources for new equipment, it is not likely that these engines will be replaced soon. How long old engines can be safely kept in service without having to change a significant portion of their structural components has been a growing concern amongst life

cycle materiel managers (LCMM). Another concern with old engines is the high rejection rates usually experienced for durability-critical parts, such as gas path components. The high replacement cost of these parts is a significant life cycle cost driver in mature engine fleets and a concern to fleet managers from an affordability point of view. The need to balance risk and affordability, has provided incentives for LCMMs to identify and implement strategies for extracting maximum usage of expensive to replace components, while ensuring the engines remain safe to operate and are reliable in service.

4.1 Engine Usage Monitoring as a Tool for Component Life Extension

One strategy for extending component lives, while ensuring safety, is to equip mature engines with modern health and usage monitoring systems (HUMS). In principle, the temporal variation of any of the engine parameters responsible for component damage can be measured with on-board monitoring systems. By applying suitable transfer functions to predict the residual lives of parts from knowledge of their operating conditions, life extension may be possible when the assumed mission severity is overly conservative.

A more precise estimate of parts life consumption obtained through the use of HUMS, in combination with an engine parts life tracking system (EPLTS), allows operators to achieve more optimal use of parts life potentials and may help them avoid in-service premature failures. In addition, engine inspection schedules and parts removal can be optimized to achieve a more cost-effective maintenance program, while ensuring safety.

However, predicting parts life consumption from knowledge of their operating conditions requires accurate models to predict rates of damage accumulation. Such models are often empirical and not very reliable. Because of related uncertainties, large safety factors are normally imposed on LCF and other life predictions. The use of more reliable damage accumulation models would preclude the need for conservatism. In this context, defining the role of environment on creep-fatigue interactions and the impact of microstructural ageing reactions of mechanical properties of engine materials is most important [1].

4.2 Other Strategies for Component Life Extension

Other component life extension options are available to engine fleet managers.

For durability-critical parts, the options include returning service-damaged engine parts to functional serviceability through use of repairs, such as welding, brazing, rebuilding, re-contouring and rejuvenation heat treatments, or delaying the rates of damage accumulation through the addition of protective coatings or surface modifications treatments. A change of material to a more durable alternative is also an option.

For safety-critical parts, the option is to implement a damage tolerance based Retirement for Cause approach for life cycle management of the components.

4.3 Implementation Requirements

The decision to replace component or extend its life by means of one of the above options, must consider the operational consequences of component failure, the cost effectiveness of the proposed life extension technique and the substantiation testing that will ensure the parts remain airworthy, safe and reliable through the life extension.

4.3.1 Safety Considerations

The associated risks depend on whether the component is functionally or structurally significant. This can be best addressed through a reliability analysis tool known as a Failure Modes and Effects Criticality Analysis (FMECA). A FMECA is a powerful tool, usually computerized, that leads fleet managers through modes of failures that can occur on components, identifies the probability of those failures and the possible consequences of failure. It provides data on component failure rates obtained from various sources (e.g. field experience, R&O, OEM, CIP, FMS), on the basis of which Fault Tree Analysis (FTA) and Reliability Block Diagram (RBD) models can be developed as management tools. A FMECA provides a basis for maintenance logistics analysis [2,18].

4.3.2 Cost Considerations

Operators must also decide on cost-effectiveness of the proposed life extension scheme for the targeted application. A cost-benefit analysis (CBA) requires that consideration be given to rates of component rejection, the cost of new parts, the cost of developing and qualifying the proposed life extension technology and the cost of applying the technology in a service environment [21,22]. A sufficient return on investment is normally required to justify implementation of any life extension scheme, unless circumstances dictate otherwise. This may happen, for instance, when no spare parts are available due to a supply shortage, or because spare parts are not produced anymore for a very old engine.

4.3.3 Technical Considerations

Implementation of any component life extension scheme outside the scope of an OEM maintenance manual, or its application beyond OEM specified damage limits, requires that the process be carried out to the same standard the original product was qualified to, or to an equivalent standard. For military aero-engines, the applicable standards are MIL-E-5007D, MIL-E-8593E (AS), MIL-STD-1783 (ENSIP) or MIL-STD-1529, although other standards may apply depending on the engine type and country of origin [17, 23]].

MIL-STD-1529 (Vendor substantiation for aerospace products) describes procedures to qualify additional/alternate vendor and fabrication sources other than those qualified for the original product. This standard is therefore particularly relevant to qualification of life extension schemes [23]. However, it does not specify technical requirements. It simply states that “substantiation tests include but are not limited to those tests required by applicable Government or engine manufacturers’ specifications”. The technical requirements are contained in these specifications.

Judging from the Critical Parts Qualification Procedures for T56 Engines [24], substantiation tests for qualifying parts subjected to life extension might consist of the following generic elements, depending on type of components and their criticality:

- (1) Dimensional inspection to ensure that the parts conform to drawing requirements;
- (2) Metallurgical verification to ensure that the materials conform to engineering specifications;
- (3) Structural tests to ensure that the relevant mechanical properties (Creep, LCF, TF,...) are equivalent to or better than properties of original parts as dictated by engineering specifications;
- (4) Functional tests to ensure parts functionality is not impaired by the life extension process (e.g. cooling flow rates for internally cooled parts are identical to flows in original equipment); and
- (5) Rig and engine tests to verify that the parts after testing still meet the serviceable limits specified in the applicable R&O Manual and that their general condition is comparable to that of approved parts subjected to an identical test.

Two types of engine test are specified in MIL-E-008593E (AS), the standard used to qualify the various models of T56 engines. These tests are:

1. Accelerated Endurance Tests (AET)

These are over-temperature tests, typically of 150 - 300 hour duration, designed to assess the durability of hot section components, wear in mechanical parts and impact of vibration of engine components. The tests are normally performed at a turbine inlet temperature (TIT) at least 8°C above maximum allowable steady state TIT for power settings at or above the maximum continuous level. The tests may be preceded and followed by a 25 hour stair-step test schedule involving rotational speed and power transients. Once the test is completed, engine performance (power, specific consumption) is compared with that of specified rating limits. The engine is then disassembled and its parts are inspected for qualification purposes

2. Accelerated Simulated Mission Endurance Tests (ASMET)

These are tests designed to simulate the effects of thermal exposure, LCF and thermal fatigue on the durability of compressor and turbine components. The tests are typically of 600 - 2000 hour duration, during which the engine is subjected to accelerated mission oriented cycles that are developed from the expected mission profile of the aircraft.

5. THE CANADIAN EXPERIENCE

The Canadian Forces (CF) have recently adopted a qualification methodology developed jointly by IAR and OAC for component life extension, which is being used to implement advanced repairs and other life extension

processes for a variety of CF aero engine parts, including components considered to be flight critical. A knowledge base and testing infrastructure has been established to support these technologically challenging tasks, with financial assistance from DND [21].

Development of the methodology started by a careful review of civil and military regulatory agency requirements applicable to the design and life cycle management of aero engines [23]. The reviewed standards included the United States Federal Air Regulations (FAR 33), the Canadian Airworthiness Regulations CAR Part V (Engine Standards - 533), the European Joint Airworthiness Regulations (JAR-E) as well as the US military standards MIL-E-5007E, MIL-E-8593A, MIL-STD-1783 (ENSIP), MIL-STD 1529 and the UK military standard Def Stan 00-971. From this review, it was apparent that the military specifications and standards are generally more stringent than civil requirements. Some standards rely more heavily on testing, including full-scale rig and engine testing. The methodology developed for the Canadian Forces is largely derived from this review. It incorporates Transport Canada requirements for Design Approval applications, which allows commercial operators to take advantage of the methodology.

The Canadian Forces Methodology for LCM of Aging Engines is an organized engineering approach consisting of:

- 1) a *Failure Mode, Effects and Criticality Analysis* (FMECA) to establish the criticality of damage found in components and its effects on aircraft performance and operation;
- 2) a *Repairability and Cost Benefit Analysis* (RCBA) to establish whether damaged components should be replaced or repaired; and
- 3) an *Engine Repair Structural Integrity Program* (ERSIP) to ensure that the parts to which a life extension process is applied will remain airworthy, safe and reliable through the life extension.

The FMECA package developed for the CF is a stand-alone reliability analysis tool designed to assist the CF LCMMs with engine maintenance. The CF FMECA identifies potential failure modes of hardware and the effects of failure on system performance, reliability and safety. It has been designed to assist CF LCMMs with identification of root failure causes and the development of corrective actions. It also identifies durability/safety critical components requiring special management approaches [2].

The Canadian ERSIP was conceived to establish structural performance requirements and identify tests for the development and qualification of life extension technologies [25]. ERSIP modifies and extends the limits of MIL-STD-1783 (ENSIP) to satisfy operator's needs for the management of components subjected to life extension processes. It incorporates the damage tolerance approach implied by ENSIP and is used by the CF to establish structural performance, process development and verification requirements that will ensure structural integrity of components subjected to life extension. The

ultimate goal of ERSIP is to ensure structural safety, durability, reduced life cycle cost and increased service readiness of engines.

The application of ERSIP involves five generic tasks, the purposes of which are summarized below [25].

Task I: Original Design and Service History Information
Calls for alignment of the proposed life extension scheme with the original structural design criteria to ensure operational needs and requirements are satisfied (may require that some design data be provided by the OEM).

Task II: Process Development and Design Modifications
Calls for coupon level qualification and quality control testing plans to develop the life extension process and any design modifications, if applicable.

Task III: Component Tests

Calls for assessing the strength, damage tolerance, durability and dynamic response characteristics of components processed for life extension.

Task IV: Engine Tests

Calls for verifying component performance in test engines either under simulated service conditions via an ASMET, or an Accelerated Mission Test (AMT) designed to simulate the life limiting damage modes of interest [2], or through field testing.

Task V: Life Management

Provides a plan for the management of repaired engines based on a life extension interval (LEI) greater or equal to the time between overhaul (TBO).

The overriding goal of ERSIP is to ensure that parts will remain airworthy, safe and reliable through the life extension. The technologies that are being addressed through ERSIP include:

- Restoration of damaged components to serviceable conditions (e.g. by repair or rework) in conformity with the original requirements, or those of ERSIP;
- Modifications intended to improve the structural durability or damage tolerance of an engine component (e.g. a material change, the addition of a coating or use of a surface treatment), in conformity with the requirements of ERSIP;
- Reuse of components under a damage tolerance-based life cycle management scheme to achieve life extension, in conformity with the requirements of ERSIP.

6. LIFE EXTENSION THROUGH RESTORATION (REPAIRS AND REWORKS)

R&O manuals contain information on some basic repairs that may be applied at overhaul to allow damaged components to be returned to service. As engines mature and service experience accumulates, it is not unusual to find out that some components are incurring damage that was not anticipated by the OEM and for which no repair procedure has been developed. Revisions to service bulletins and manuals for military engines are issued from time to time by OEMs to introduce new repairs or changes in design and materials with a view to addressing component durability shortcomings. This is most often

done under a Component Improvement Program (CIP). However, the introduction of new repair or rework procedures by the OEMs often falls behind advancements in technology. The latter provide opportunities to create new repairs or implement new design or material changes, such that components experiencing unforeseen deterioration can be returned to service, instead of being scrapped [26]. New technologies may also allow parts to be repaired beyond the damage limits envisaged and set by the OEM as given in the standard R&O Manuals.

Much work has been done in the commercial world to develop repairs and reworks for civilian aircraft engines. Repair vendors have been competing quite successfully with OEMs in these developments. The delegation of authority by National Aviation Authorities to R&O Organizations has encouraged such initiatives. The technology developed for civilian products is for the most part applicable to military platforms and can be adopted by military organizations to achieve cost-effective management of their engines [2,26]. The following listing identifies types of repair/rework technologies that CF engine fleet managers have considered in order to achieve component life extension:

- New welding materials and techniques for cracked and worn parts;
- Advanced brazing or diffusion bonding techniques for cracked components;
- Rejuvenation of properties in creep or fatigue damaged parts by HIPing and/or heat treatment; and
- Surface rebuild or re-contouring for eroded, corroded or worn parts.

6.1 Advanced Welding Repairs

Welding technologies have long been used to repair cracked gas path components such as vanes and blades and combustor liners. Procedures are well established and detailed in the engine R&O Manuals. However, the emergence of new techniques makes it possible to repair components that are not covered by the standard manuals or beyond the limits set by the OEM [1].

For instance, OAC has recently developed a patch repair for eliminating FOD in F404 fan and compressor blades using electron beam welding (EBW) [27]. The damaged area is replaced by a patch made of the same material as the blade, which is joined to the airfoil by EBW. The repaired assembly is then finished-machined to meet dimensional specifications, Fig 13. These blades have been shown to perform as well as new blades in qualification tests and will shortly be field-tested in a CF engine. A similar process is used at Pratt & Whitney Canada (PWC) to repair airfoils of integrally machined titanium alloy impellers and blisks [9].

The substantiation tests used to develop and qualify the weld repaired fan blades included evaluating the microstructure, tensile strength and fatigue properties of welded test coupons as well as comparing properties of new original equipment with properties of repaired parts. The comparison included vibration characteristics,

fatigue properties and impact resistance of the components. The instrumented ballistic test rig developed to compare the resistance to impact by a high velocity projectile of new and weld repaired fan blades, is shown in Fig 14. The weld-repaired components were finally subjected to testing in an engine, as part of a CF F404 accelerated mission test (AMT) performed on behalf of the CF at IAR. This AMT was used to qualify the fan blade repair as well as several other life extension schemes developed by OAC and IAR.

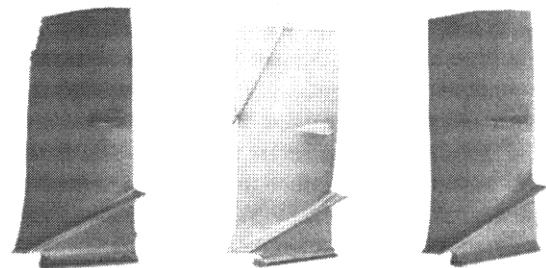


Figure 13. Repair of FODed F404 fan blade by electron beam welding a corner patch to replace damaged portion of blade.

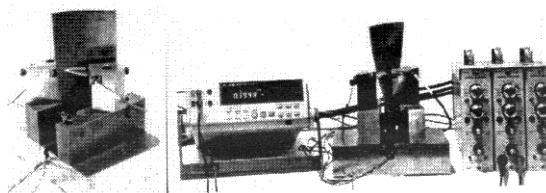


Figure 14 Instrumented ballistic test rig for assessing resistance to impact by high velocity projectile of EBW repaired fan blades.

Techniques such as Dabber™ welding [28], or powder metallurgy welding techniques [29], as well as pulsed laser or plasma welding techniques are making it possible to weld high strength superalloy components that were previously difficult to weld. Automated welding systems, in combination with laser or micro-plasma power sources, have allowed the application of weld metal consistently, using extremely low heat input to minimize effects on the base material [10].

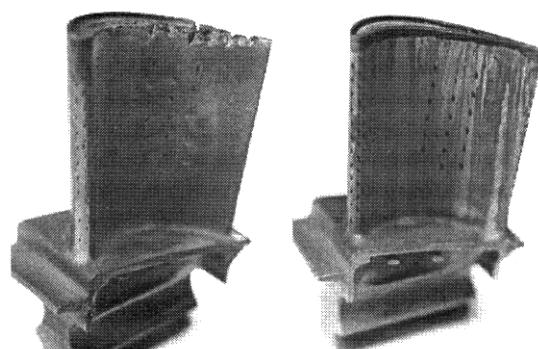


Figure 15. Weld tip repair of DS René 80 first stage turbine blade form CF F404 engine (Courtesy Liburdi Engineering, Hamilton, Ontario)

The tip repair of a F404 directionally solidified first stage high-pressure turbine (HPT) blade is illustrated in Fig. 15. Here, the tip oxidation/cracking damage was first removed by grinding. The blade tip was then rebuilt by vision assisted automated plasma arc welding, using weld material chosen to provide enhanced oxidation resistance [10].

The repair of worn seal teeth from rotating air seal components is another economically desirable refurbishment procedure. The knife-like circumferential seal teeth on the outer surface of air seal components that intrude into a surrounding abradable shroud, are susceptible to wear during service. The teeth also get damaged during assembly and disassembly at overhaul. The tips of the damaged teeth are first ground down and then rebuilt with over-lays of similar alloys, using welding techniques such as Dabber™ [28] welding or pulsed laser or plasma torch welding [10]. An example of refurbished seal teeth prior to final machining to the desired shape is shown in Fig. 16.

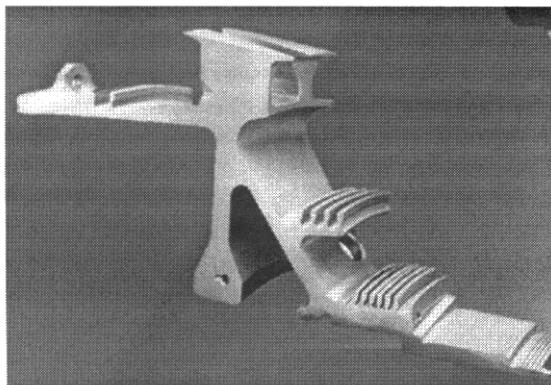


Figure 16. Example of seal teeth repair using the Dabber™ technique (Courtesy of Liburdi Engineering, Hamilton, Ontario).

Substantiation tests and analysis for weld repair of turbine hot section components may be quite elaborate. For the seal teeth repair for instance, it is necessary to assess the risk of cracks initiating at imperfection in the built up weld or at the weld metal/parent metal interface. The presence of undetected cracks at these locations may cause a significant debit in LCF properties of the component. Test results reported by Domas [28], obtained with simulated seal teeth specimens, in combination with fracture mechanics analysis, provided an insight into the likely performance of repaired components, and contributed to the establishment of process capability and margins in relation to specification limits.

6.2 Advanced Brazing Repairs

The development of hydrogen fluoride (HF) based cleaning processes such as the AFOR-DBR of Vac-Aero International or the University of Dayton Research Institute (UDRI) oxide reduction process, in combination with diffusion brazing technology, has made it possible to repair previously unbrazeable nickel base superalloys. These processes rely on gaseous HF to remove

thermodynamically stable oxide from crack faces, thus promoting wetting and infiltration of the braze alloy by capillary forces and, thereby, creating a structurally sound and durable joint, Fig 17 [10, 30].

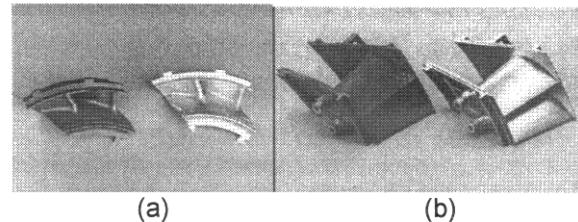


Figure 17. (a) INCO 792 compressor turbine test pieces in the as received condition and after AFOR/DBR processing (Courtesy of Vac-Aero International); (b) HF cleaned and braze repaired René 80 LPT nozzle (Courtesy of Orenda Aerospace Corporation).

The Liburdi Engineering LPM™ joining/cladding process [10] was developed as a hybrid wide gap brazing technique that has proven successful for the repair of both blades and vanes. The process enables a wide range of alloys to be used for crack repair and surface build-up. The damage is first removed by grinding and a powder metallurgy putty of matching or custom composition is applied and diffused into the surface to complete the repair and achieve the desired mechanical and metallurgical properties.

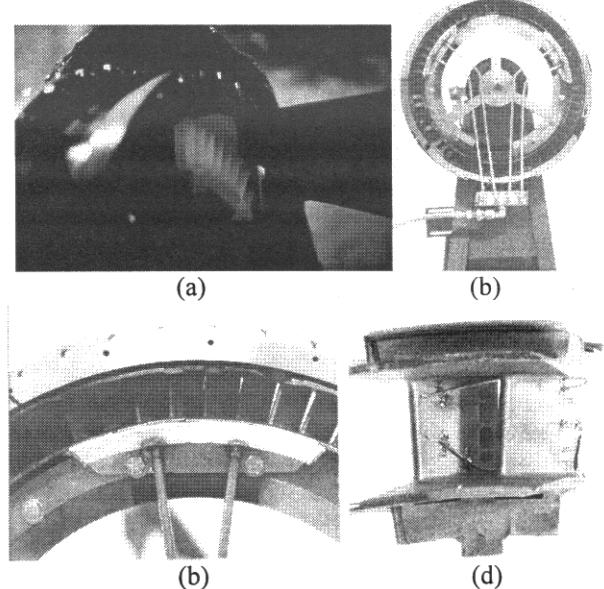


Figure 18. Hardware for thermal fatigue testing of NGVs in burner rig: (a) and (b) specimen fixture and cooling arrangement; (b) close-up of specimens mounted in fixture; (c) fine wire thermocouples spot welded on air foils for temperature calibration

Substantiation tests for repairs of turbine hot section components would typically include an evaluation of the microstructure and the mechanical properties (stress rupture, LCF) of welded or brazed joints, using test coupons designed to simulate joint geometry and loading conditions applicable to the repaired part. Rig tests would

also be performed on components to ensure for instance that the flow rates through internal cooling passages are identical for new and repaired parts. The effective flow area of a nozzle guide vane is another parameter that would have to be checked for compliance to specifications. Other types of rig tests may be considered to compare the durability and damage tolerance of new and repaired parts under simulated service conditions [31]. Finally, the repaired parts would be subjected to engine testing and/or field evaluation [2].

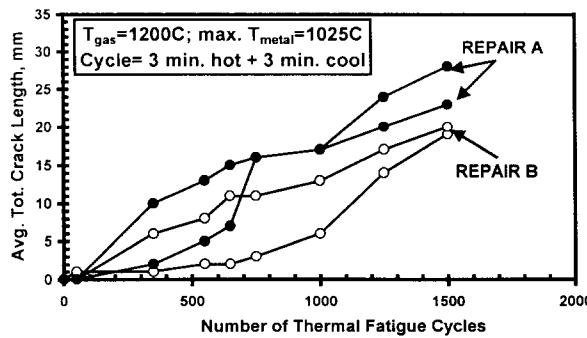


Figure 19. Crack growth rate data during burner rig testing of NGVs repaired by two different processes [31].

At NRC-IAR, engine specific test hardware has been developed for use in a high velocity burner rig to compare the thermal fatigue response of repaired NGVs when subjected to thermal cycling in high velocity gas jets (Mach 0.3 to 0.8). Elements of the hardware used to assess the durability and damage tolerance of repaired T56-A15/A14 first stage NGVs are shown in Fig 18. The burner rig is a Becon LCS-4B combustor fitted with an exhaust nozzle that simulates the geometry of the actual nozzle of the engine combustor. The parts to be tested are mounted in front of this exhaust nozzle, in a wheel assembly made up of an entire set of NGVs, Fig 18(a). During testing, the assembly is rotated back and forth by a pneumatic actuator to alternately expose two sets of test components to the rig hot gases and a cooling air jet at room temperature. Internal cooling of the test components can be provided, if desired. For T56 engines, three NGV segments are tested simultaneously. While one set of test components is being heated in the hot gas jet, the other set is being cooled in the air jet. The rotation of the wheel assembly occurs at predetermined time intervals to ensure that steady state temperature is achieved during both the heating and cooling portions of the cycles. Hot gas temperature and rate of cooling airflow are adjusted to achieve component surface temperatures that are close to the operating temperatures of the parts. The test is interrupted after predetermined numbers of thermal cycles to inspect the NGVs for the presence and size of cracks. Crack growth rate data obtained for two sets of T56-A15/A14 first stage NGVs repaired by different vendors are shown in Fig 19 [31].

6.3 Rejuvenation by Heat Treatments

Heat treatments and hot isostatic pressing (HIPing) have been used for over two decades to eliminate service induced microstructural damage in turbine blades and

vanes to restore creep properties [1]. HIPing eliminates creep voids that may have formed during service. The re-coating heat treatment completes the rejuvenation. HIPing also eliminates casting porosity, which improves component reliability by reducing scatter in material properties. HIP rejuvenation cycles have been developed at IAR for alloy 713C and IN 738, both of which are used in T56 engines [32,33]. HIPing can also be used in conjunction with other forms of repairs, for instance to eliminate shrinkage porosity within braze joints [30].

The qualification of HIP rejuvenation treatments requires that mechanical properties of rejuvenated components be established from tests performed on specimens machined from components. Miniature creep specimens machined from T56 turbine blades are shown in Fig. 20(a). The rupture life and creep elongation data shown in Fig. 20(b) for new, service-exposed and rejuvenated IN 738 blades indicate that loss of creep ductility induced by service can be fully recovered by HIP rejuvenation, while time to rupture relative to new blades can actually be improved, providing post HIP heat treatments are carefully designed [33].

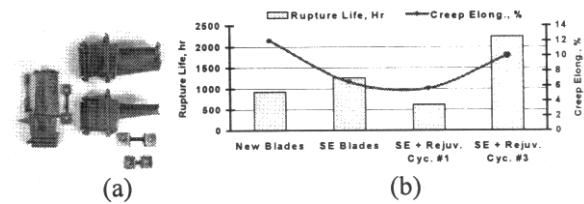


Figure 20 (a) Miniature creep specimens machined from the airfoils of different T56 turbine blades; (b) Comparison of rupture life and creep elongation of new, service-exposed and HIP rejuvenated blades showing influence of HIP cycle and improvements of properties achieved with cycle #3 [13,31]

6.4 Re-contouring of Compressor Airfoils

Processes such as Sermatech's RD-305 or Lufthansa/University of Aachen's ARP can be used to recover the loss in aerodynamic performance of eroded compressor airfoils. With these processes, the leading edge of eroded airfoils is re-machined by grinding to a new optimized contour geometry designed to meet the flow condition characteristics of the original blading. The design criteria are chord length, profile thickness and leading edge angle. Re-contouring is limited to blades for which the chord length is greater than the OEM specified minimum allowable, and therefore the components are deemed reusable. Re-contouring translates into increased compressor efficiency, reduced fuel consumption, extended on-wing times and reduced spare parts costs [34].

7. LIFE EXTENSION THROUGH MATERIALS MODIFICATIONS

The options available to enhance component durability through materials modifications include:

- Substituting a component material for another, such as replacing a blade alloy from a conventionally cast

form to a DS product (CG or SC) to increase the component creep strength or its thermal fatigue (TF) resistance;

- Replacing or adding a protective coating to improve resistance to wear, fretting, erosion, oxidation or corrosion, such as hard coatings to protect compressor airfoils against erosion or a TBC for hot parts; and
- Applying a surface modification treatment such as shot peening, ion implantation or laser surface treatments to improve resistance to various modes of surface degradation or fatigue.

7.1 Retrofitting with New Materials

The decision to change a component material is normally made during a component improvement program led by the engine manufacturer but supported by different user services. There are examples of material changes involving both blades and discs. For CF F404 engines, the first stage HPT blade material was changed from CC René 125 to DS-CG René 80 and, recently, a change to SC alloy N4 has been approved. This was done to improve the creep strength and thermal fatigue resistance of the blades. However, with changes of this type, the expected improvements are not always met, because a different mode of damage that was not anticipated may prove to be life limiting. Disc alloys are more rarely changed. Any change would be made to correct a durability shortcoming. For example, one of the CF J85 CAN40/15 engine compressor disc was changed from AM355 martensitic stainless steel Fig. 9(b) to a DA718 nickel base superalloy to eliminate premature bolt hole cracking.

7.2 Replacing or Adding a Protective Coating

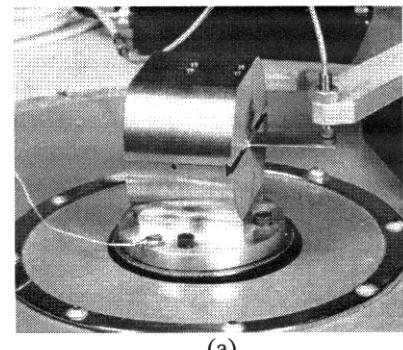
Both the substitution and the addition of coatings provide opportunities for life extension of cold end as well as hot end components. The qualification testing would typically involve documentation of the physical and mechanical properties of the new coating as well as evaluation of relevant durability and damage tolerance properties. More importantly, coating-substrate interactions and their impact on component properties would have to be carefully assessed.

7.2.1 Hard Coatings for Compressor Airfoils

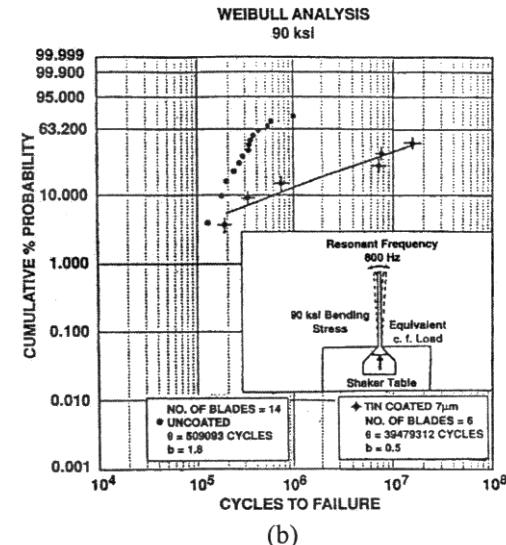
Titanium nitride (TiN) has been qualified as a coating for the protection of compressor airfoils against damage due to erosion and corrosion. In particular, the RIC™ PVD TiN coating from Liburdi Engineering is a bill-of-material option for T56/K501 RR(Allison) engines [10]. Titanium nitride is a hard ceramic compound, which provides excellent protection against erosion and some corrosion protection, although pitting may be a problem with some commercial TiN coatings. The latest generation of coatings are designed to improve both erosion and corrosion resistance. These new coatings are based on alloying TiN with carbon or aluminum (TiCN, TiAlN). Coatings based on chromium nitride are also being considered for corrosion protection [5].

Substantiation testing for compressor coatings will typically involve measurements of coating thickness, uniformity and surface finish, assessment of coating durability (adhesion, hardness, ductility and toughness, erosion resistance, corrosion resistance). The effects of coating on substrate microstructure and mechanical properties (strength, fatigue), on vibration characteristics and fatigue properties of coated components, and on compressor efficiency and engine performance would also be considered [5,31]. A sand ingestion test may be conducted inside a test cell. However, because such testing may not accurately simulate real operating conditions, a field evaluation is usually preferred.

Coatings can significantly alter the natural frequencies of compressor airfoils. The change may be sufficient to cause the airfoils to be excited at resonant frequencies during service and such vibration may lead to high-cycle fatigue (HCF) failure. The effect of coatings on the fatigue life of the component is another important consideration of the qualification process. Substantiation test hardware developed at IAR to assess the effects of coating on fatigue properties of T56 6th stage compressor blades is shown in Fig. 21(a) [31].



(a)



(b)

Figure 21. (a) Apparatus for measuring natural frequencies of fatigue properties in first bending mode vibration of 6th stage T56 compressor blades; (b) Weibull distribution plot of fatigue life of the coated and uncoated blades measured with the apparatus [5].

The blade is attached by its dovetail to a fixture mounted on an electromagnetic shaker and made to vibrate at its natural frequency in the first bending mode, until failure occurs. The number of cycles to failure of coated blades is compared with that for bare blades at identical root stress levels. Weibull analysis of the fatigue test results for RIC coated blades indicates that the cumulative probability of failure for the coated blades is significantly smaller at 90 ksi than for the uncoated blades, Fig. 21(b) [31]. The S/N data obtained by rotating bending fatigue also indicate that RIC coated test coupons of the blade material (17-4PH stainless steel) have fatigue lives comparable or better than those of bare coupons [5]. The increase in fatigue life may be due to changes in the blade microstructure induced by the coating process or to the residual compressive stresses induced by the coating on the surface of the blades.

7.2.2 Coatings for Turbine Hot Section Components

It is common practice to re-coat turbine blades and vanes at overhaul. This provides an opportunity to change the coating for one that is better suited for the operating environment. For instance, a Pt-aluminide will provide some durability enhancement over conventional aluminides where hot corrosion is found to be life-limiting. Also, the addition of a thermal barrier coating (TBC) to some critical areas of an airfoil, such as the tip region of turbine blades or leading edge of an NGV, will lower component metal temperature and minimize the rate of material consumption due to oxidation at these locations [2]. TBCs will also reduce the sensitivity of these components to thermal fatigue cracking by decreasing the transient stresses induced by thermal cycling during service [2].

Coatings can also be added to components that have been designed to operate without a protective coating. This is the case for the F404 turbine nozzle. Rapid oxidation of the airfoil surface of this component gives rise to loss of material through spallation and thermal fatigue cracking near cooling holes. OAC has developed a braze repair for the cracks and a repair-compatible coating for the airfoil that has been shown to greatly enhance component durability in accelerated burner rig tests [2,35].

Substantiation testing to qualify new protective coatings for turbine hot gas path components will typically involve a metallographic evaluation of coating microstructure to evaluate quality of the coating and its impact on microstructure and phase stability of the substrate material. An assessment of the effects of coating on the mechanical properties of coated material test coupons (Tensile and creep strength, HCF LCF, thermal mechanical fatigue (TMF), ductile-brittle transition temperature (DBTT), etc.) is another important requirement. Rig tests are required to ensure that the flow rates across cooling passages of internally cooled parts are identical for original and modified parts. Rig tests are also required to compare the durability and damage tolerance of original and modified parts, under simulated service conditions. Engine testing as part of an accelerated mission test (AMT) will complete the qualification. The IAR burner rig is used to assess the

durability of modified components as well as to screen candidate coatings applied to test coupons in the form of pins or plates, depending on the type of coating and intended applications [31].

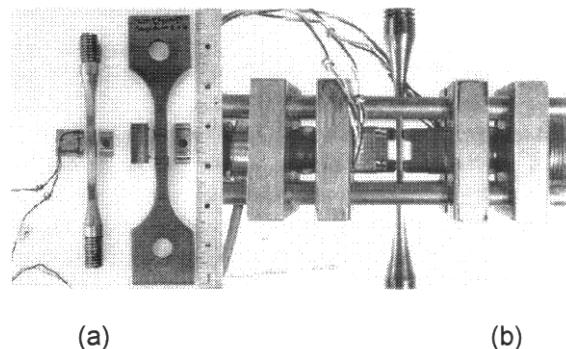


Figure 22. (a) Tensile coupons and (b) IAR fretting fatigue test rig for assessment of fretting fatigue resistance of fan engine materials [31].

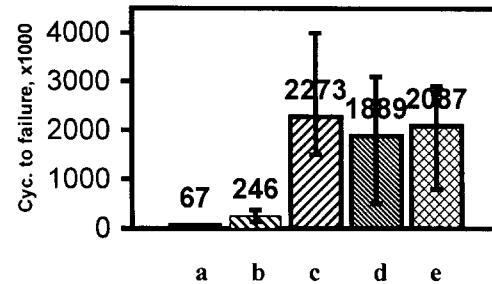


Figure 23. The fretting fatigue life of Ti-6Al-4V:
a = base metal; b = CuNiln + MoS₂; c = shot peened; d = shot peened + CuNiln + MoS₂; e = shot peened + MoS₂.

7.3 Surface Modification Treatments

Surface treatments such as ion implantation and chemical surface treatments have been explored for use in conjunction with shot peening and soft coatings to alleviate the fretting fatigue problems in the dovetails areas of titanium alloy fan and compressor blades. Shot peening in combination with chemical treatments has been found to be quite effective in laboratory tests but other surface treatments, including soft coatings and lubricants, appear to also have the potential to improve fretting fatigue resistance of titanium alloys. The effects of these treatments on fretting fatigue life of two titanium alloys (Ti64 and Ti17) have been evaluated using an apparatus developed at IAR, Fig. 22. Some of the results of this work are presented in Fig. 23 (36,37).

8. LIFE EXTENSION OF SAFETY-CRITICAL PARTS THROUGH IMPLEMENTATION OF NEW LCM METHODOLOGIES

The main factors considered in the choice of materials for compressor and turbine discs in engines developed in the 50's and 60's were tensile properties in the bore and the rim of the discs and creep properties in the rim of turbine discs. This was done essentially to provide an overspeed margin without disc burst. For engines of that generation,

fatigue lives were not provided for safety-critical parts [38]. On occasion, LCF lives of turbine rotors have been established by OEMs at a late stage in the life of an aging engine, usually under a component improvement program (CIP).

For a number of such undertakings, the actual service lives of a significant fraction of components from lead-the-fleet engines have been found to be greater than the calculated LCF safe lives. This reinforces the view that components retired at their design safe-life limits may have significant fractions of usable life remaining. This is the case for an engine operated by the Canadian Forces. In another situation affecting the Canadian Forces, the engine was old enough that replacement discs and impellers were not available. In situations of this type, the implementation of a damage tolerance approach can assist fleet managers minimize risk if they must keep the engines in service.

Table 1 CF engine components targeted for life management based on RFC concepts and status of the analyses.

Engine	Component	Status
Nene X	Impeller	Complete
	Turbine disc	Complete
	Turbine shaft	In progress
J85	Compressor Disc	Complete
	Turbine disc assembly	In progress
T56	Turbine spacer	In progress

Implementation of a damage tolerance based LCM approach for safety critical parts is best undertaken by the OEM, for instance under CIP sponsorship. However, there may be situations where this is not possible, either because CIP sponsorship is not an option or the OEM is not prepared to undertake the required work.. In this case, there may be no alternative but for the user to undertake the implementation.

In Canada, components from three CF engines have been targeted for implementation of a damage tolerance/RFC based LCM approach. The components are from Nene X, T56 and J85 engines and they are identified in Table 1, along with the status of the undertakings.

Implementation of a damage tolerance based LCM methodology for aging components requires information on (i) initial crack length, (ii) changes in stress intensity factor (SIF) with crack extension, (iii) fatigue crack growth rates (FCGR) at the fracture critical location in the component, (iv) dysfunction crack size (DCS) and (v) cyclic and/or steady state service usage of the engine. This information is needed to predict a fracture mechanics based safe inspection interval under typical service conditions. There are seven steps that would normally be followed to obtain or generate the required information and to conduct and validate the analyses.

The seven steps include:

- Determination of stress and temperature data for the component of interest;
- Identification of the fracture critical location in the component;
- Determination of the stress intensity factor (SIF) of cracks at the fracture critical location and its dependence on crack size and estimation of the dysfunction crack size (DCS);
- Generation of fracture mechanics data for safe inspection interval (SII) calculations
- Generation of POD data for the NDI technique used at depot level to inspect the component (to provide estimate for initial crack size);
- Calculations of the safe inspection interval (SII); and
- Full scale testing in a spin rig to validate the SII.

Examples taken from CF experience for each of the steps are presented below.

Step 1: Determination of Engine Stress and Temperature Data

Mechanical stresses can be calculated from rotational speed, geometry and mass of the component. However, predicting thermal stresses requires detailed information on temperature transients and temperature maxima, which are more difficult to obtain. This information is strongly influenced by service usage and may not be available from the engine manufacturer. For the J85 turbine disc assembly, work is underway at IAR to generate the information under the auspices of a TTCP project. A turbine assembly has been instrumented to determine the temperature distribution across the components. An engine has also been equipped with a slip ring to transmit the information collected from sensors to instrumentation located outside the engine. A photograph of the modified engine is shown in Fig. 24. The results of this study will be used to set boundary conditions in FEM models to establish temperature and stress distribution within the turbine assembly.

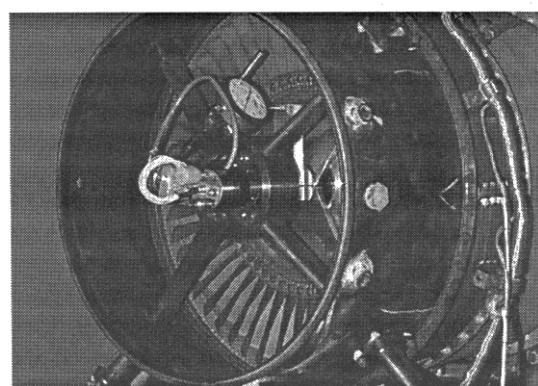


Figure 24. Back end of J85 CAN40 engine with slip rig installed to transmit sensor information to an outside data acquisition system.

Step 2: Identification of the Fracture Critical Location

This is obtained by analysis using the finite element method (FEM). For Nene X components, the thermal-

mechanical loading information required as boundary conditions was discussed with the engine manufacturer. Both 2-D and 3-D analyses were performed for the turbine disc. The bottom serration of the firtree slots in the disc was identified as the fracture critical location because of a combination of high thermal and mechanical stresses operative at this location [39]. The distribution in von Misses stress between two of the disc slots is shown in Fig. 25 for the 2-D analysis case.

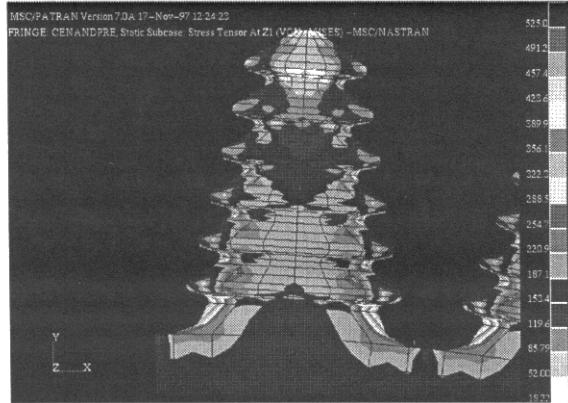


Figure 25. Details of the 2D stress distribution for the rim region of the Nene X turbine disc in an uncracked condition

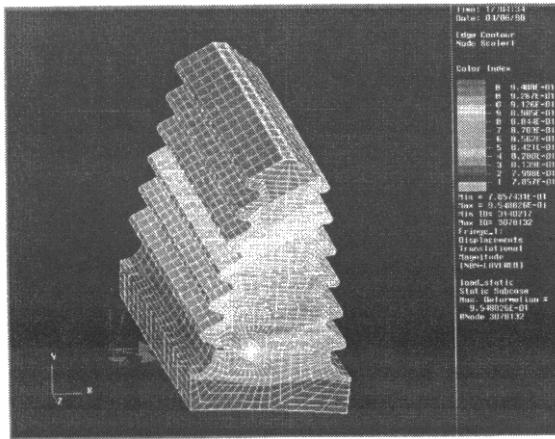


Figure 26. Details of the 3D strain distribution for the rim region of the disc with a modeled crack in the bottom serration of the firtree slot, which is the primary fracture critical location in the disc [39].

Step 3: Fracture Mechanics Analysis for SIF Calculations

This is also obtained analytically using the FEM. For the case of the Nene X turbine disc, a crack was assumed to be present in the bottom serration region of the firtree slot, as shown in Fig. 26. The fracture mechanics analysis was performed for cracks with surface length to depth ratios ($2c/a$) of 3:1 and 4.5:1. A $2c/a$ ratio of 3:1 is often observed in aero engine discs. A $2c/a$ ratio of 4:1 is considered to represent a worst case situation, whereas a $2c/a$ ratio of 4.5:1 represents a multiple crack situation [39]. Variation of the stress intensity factor (SIF) with crack depth for a crack with a $2c/a$ ratio of 4.5:1 is shown in Fig. 27 [40]. For a crack 8 mm deep, the SIF value ahead of the crack tip is 64 MPa $m^{1/2}$, which is a fraction

of the fracture toughness (K_{Ic}) of the material. That crack depth magnitude of 8 mm was used in all subsequent calculations to arrive at the SII for this component.

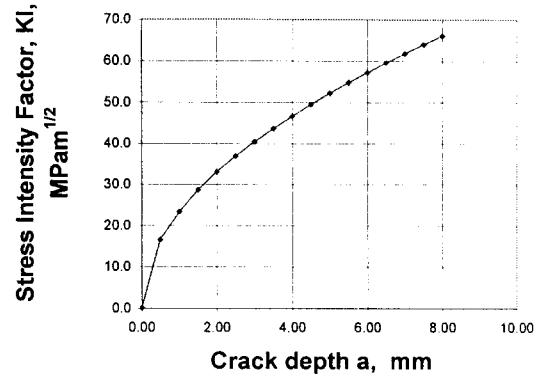


Figure 27. Stress intensity factor (SIF), as a function of crack depth for a crack with a $2c/a$ ratio of 4.5:1 located in the bottom serration of the firtree slot in the Nene X disc [40].

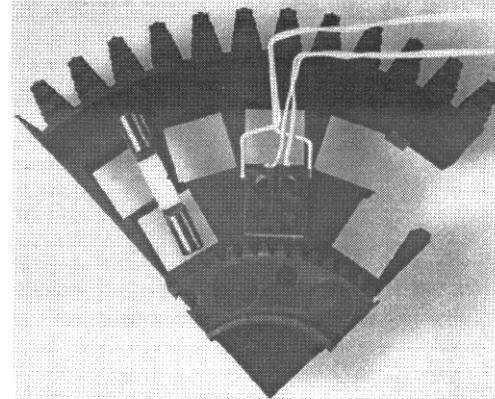


Figure 28. Tensile and compact tension specimens machined from the Nene X disc. The electrical leads are part of the automated DC-PD technique used to monitor crack growth during testing.

Step 4: Generation of Fracture Mechanics Data

The fatigue crack growth rate (FCGR) data versus stress intensity range, ΔK , is preferably generated using compact tension specimens machined from a high time component as close as possible to the fracture-critical location, Fig. 28. A high time component is selected to account for any effects of service-induced microstructural degradation on crack growth. The test temperature is chosen on the basis of service temperature estimates for the fracture critical location. The specimen geometry should conform to ASTM E-647 specifications to ensure that plane strain conditions prevail ahead of the crack tip during testing. The FCGR data reproduced in Fig. 29 reveals that material machined from a high time IN 718 disc has significantly higher FCGRs than material machined from a new or a medium time disc (41). Such variations in FCGR data would significantly influence SII calculations. The worst case scenario would normally be adopted for the calculations [42,43].

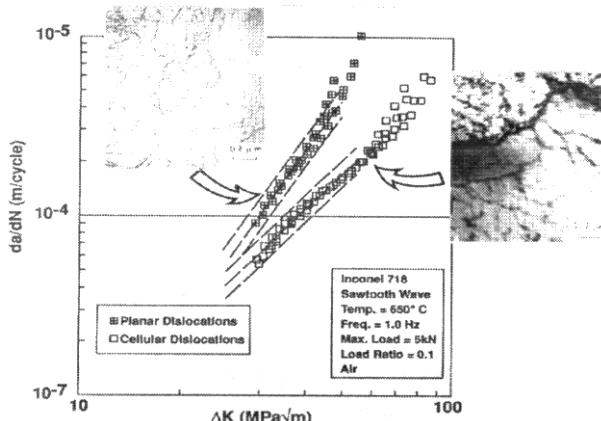


Figure 29. Comparison of fatigue crack growth rates (FCGR) in new and service exposed alloy 718 turbine discs; high time disc shows significantly higher FCGR as compared to new and medium time disc [41].

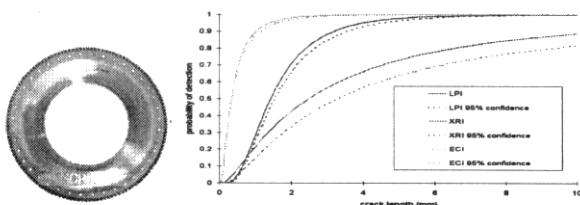


Figure 30. Probability of detection of natural cracks in Fe-Ni-Cr alloy turbine discs using liquid penetrant inspection (LPI), eddy current inspection (ECI) and x-ray inspection (XRI) [44,45].

Step 5: Generation of POD Data for the NDI Method of Interest

The size of cracks likely to be missed during component inspection needs to be estimated to predict SII's. Ideally, enough data should be generated to obtain probability of detection (POD) curves specifically for the component of interest and the method used for its inspection. This may not always be possible. The initial crack size (a_i in Fig. 11) for components targeted by the CF have been selected from a demonstration program carried out in Canada on a series of J85 life expired compressor discs containing natural fatigue cracks originating from bolt holes [16]. For this program, several discs were inspected by a variety of NDI methods and by different inspectors. The NDI results were verified by prying open each bolt hole inspected and examining the fracture surface for evidence of service induced cracking. Actual flaw sizes were established from the microscopic examinations. Finally, POD data for each of the NDI methods considered was generated through standard POD analysis. This program was first introduced to the AGARD community in 1988 as part of a Workshop on Non Destructive Evaluation [44]. A follow-up AGARD sponsored activity broadened the scope of the study for the benefit of the NATO community [45]. The substantial data on POD of NDI methods generated through this effort are available from the USAF supported

Nondestructive Testing Information Analysis Center (NTIAC) in Austin, Texas. The POD curves for three common inspection techniques, including liquid penetrant inspection (LPI), are compared in Fig. 30.

For the Nene X turbine disc, the initial flaw size was assumed to correspond to either the maximum crack length missed (a_{max}) or the crack length value at 90% POD with 95% confidence (90/95 POD) obtained for LPI under the J85 disc program. This is justified since LPI is used to inspect Nene X components and the two discs are made of stainless steel [40].

Step 6: Calculation of SII

The SII is set as a fraction of the time it takes to grow a crack of assumed geometry from the initial crack size to the dysfunction crack size. Either deterministic or probabilistic fracture mechanics (DFM or PFM) analyses can be used to compute the SII. A DFM analysis is based on worst case situations for all variables in the life calculations. Many have claimed DFM to be overly conservative and this explains the growing interest in PFM based analyses. A PFM analysis aims primarily to simulate the consequences of missing a crack during inspection (42,43). It also helps account for scatter in FCGR data.

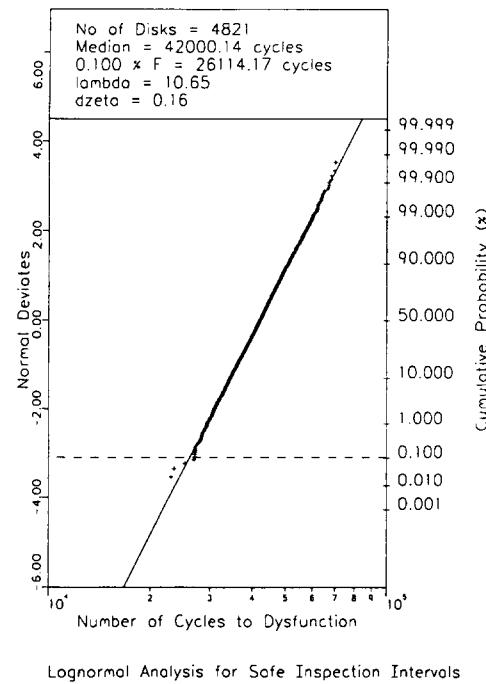


Figure 31. Lognormal analysis of PFM generated data for Nene X turbine discs simulating the effect of worst possible scatter in FCGR and uncertainties associated with the LPI technique [40].

A lognormal analysis of PFM data generated for the Nene X turbine disc, which simulates the effect of worst possible scatter in FCGR and uncertainties associated with the LPI technique is presented in Fig 31. The data reveal that the 0.1% probability of failure or 1 in 1000 chance that a missed crack will reach dysfunction size is of the order of 26,000 cycles. Assuming the worst case

scenario of 5 cycles per hour of engine usage, the 0.1% probability of failure translates into 5000 hours. Since the engine is currently inspected every 1800 hours, it is not surprising that no components have ever failed catastrophically in service. Interpretation of PFM analysis simulating experimentally observed scatter in FCGR data indicates that the overall safety factor inherent in the predicted SII for the fleet size is 4.7 [40].

Final decisions about setting SIIs for the Nene X disc were made on the basis of a DFM analysis. A worst case crack propagation curve (depth versus cycles) for a crack 1 mm deep originating from the fracture critical location and growing to a DSC depth of 8 mm is shown in Fig. 32. The data clearly demonstrate the importance of the sensitivity of the NDI technique on the SII. For instance, if the initial crack depth is 1 mm the crack propagation interval (CPI) is about 30,000 cycle, whereas it reduces to about 8,000 cycles if the detectable crack depth is only 4 mm. The SII recommended for the Nene X disc is 1/3 of the CPI, or 10,000 cycles. For the worst case scenario of 5 cycles per hour of engine usage, the DFM based SII expressed in flying hours is 2000 hours. Therefore, there is high probability that any crack would be detected during routine overhaul inspection, since the routine inspection for this engine is conducted every 1800 hours [40].

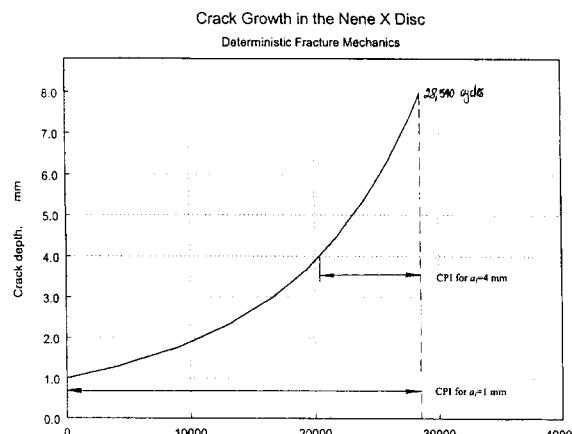


Figure 32. Damage tolerance based life cycle management curve for Nene X turbine discs using 90/95 POD value of the LPI technique as the detection limit of the technique [40]

Step 7: Validation of SIIs Through Spin Rig Testing
Validation of the SII calculations requires spin rig testing of components under conditions that simulate service. Testing of J85 and Nene X components is underway at IAR to validate component lives and safe inspection interval calculations using the spin rig facility shown in Fig 33. To date 9,000 mission cycles have been accumulated on the J85 turbine disc assembly at a test temperature of 450°C. Both LPI and eddy current inspection techniques are being used to inspect the components as the cycles accumulate.

All above predictions are based on the assumption that cyclic usage was the primary driver for crack initiation

and growth processes in the Nene X disc. This assumption was made on the basis of discussions with the OEM, from which it was also concluded that time dependent creep mechanisms did not contribute to the crack growth process to any significant degree during service. Based on past experience with the fatigue response of turbine disc materials, it was also assumed that LEFM could be used to describe crack growth within the component. Had creep contributed to the accumulation of damage in any significant way, creep-fatigue interactions would have had to be considered.

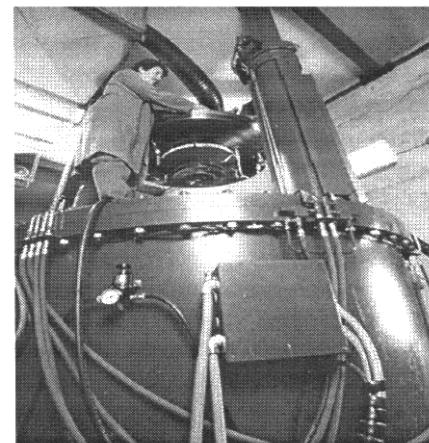


Figure 33. J85 turbine disc assemble being readied for testing in IAR spin rig.

CONCLUDING REMARKS

Aging in engines is a process involving gradual deterioration of components, which begins when an engine enters service. This aging process cannot be avoided and must be managed. Managing the aging of engine components to extract maximum usage of expensive to replace components requires a good understanding of deterioration modes and their effects on engine performance, reliability and safety. From an operator's perspective, a number of options are available to extend component lives beyond book limits. The options include taking advantage of technology developments to repair damaged components or enhance their durability through material modifications, including the substitution or addition of protective coatings. The implementation of a damage tolerance based life cycle management methodology for safety-critical parts may also be a practical option in some situations. The Canadian experience suggests that significant cost savings may accrue from such undertakings [21].

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AGING AVIONICS- A SCIENCE & TECHNOLOGY CHALLENGE OR ACQUISITION CHALLENGE?

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The fleet of aircraft which we have today, represents 90% of the aircraft which we will have well into the next century. With defense budgets shrinking, fewer new aircraft will be purchased. Avionics modernization is the highest leveraged approach to maintain and increase capability. This treatise discusses the problems associated with our existing fleet of aircraft, both from a maintenance and modernization perspective. Examples of obsolescence and retrofit issues are provided to highlight the seriousness of the problem.

The science and technology challenges are the focus of a small set of technologies aimed at reducing the cost of our aging systems. These costs include maintenance as well as upgrade costs. Aging systems are different from new systems because there is an existing infrastructure. It is much easier to start from "scratch" than modify something already in existence. Specific technology areas that will be discussed include:

- a. Avionics software
- b. Group A Modification
- c. Avionics Maintenance
- d. Application of COTS products
- e. Obsolescence/Environment for Rapid Design Changes

Acquisition related issues have to be addressed in consonance with the Science and Technology challenges. Issues that will be discussed include: Open systems, planned technology insertion/planned obsolescence, verification/validation, warranties/repair strategies and organic/contractor support. As part of this acquisition discussion, industry pursuit of a product line investment strategy will be addressed as their adoption of best commercial practices continues.

A possible Avionics S&T strategy will be discussed which outlines an approach for infusion and transition of advanced technologies into the existing fleet given an acquisition friendly environment. Discussion on modeling, simulation, and prototyping will be provided to further explain the fleet wide application of these technologies.

Background

Today DoD is being challenged to do more with less. This has focused attention on affordability opportunities and challenges with emphasis on commercial or commercial-based avionics and electronics. The life of older (legacy) aircraft is being extended while their avionics systems are becoming obsolete, and more difficult and expensive to maintain and support. New aircraft also need to be more affordable and have functionality and performance equal to or exceeding that of existing systems. Lower upgrade and support costs for existing aircraft and lower acquisition and support costs for next generation avionics systems, are primary challenges.

Our Nation's aircraft are capable, but their electronics are old and aging rapidly. In the post 2000 era, these aircraft will experience worldwide deployment in support of our active front-line forces. The majority of our Nation's fleet of aircraft through 2020 will be legacy aircraft as shown in Figure 1.

The tactical fighter fleet is aging rapidly in large numbers, with the projected average age of all fighters in the USAF and Navy inventories estimated to be over 20 years by the year 2020.

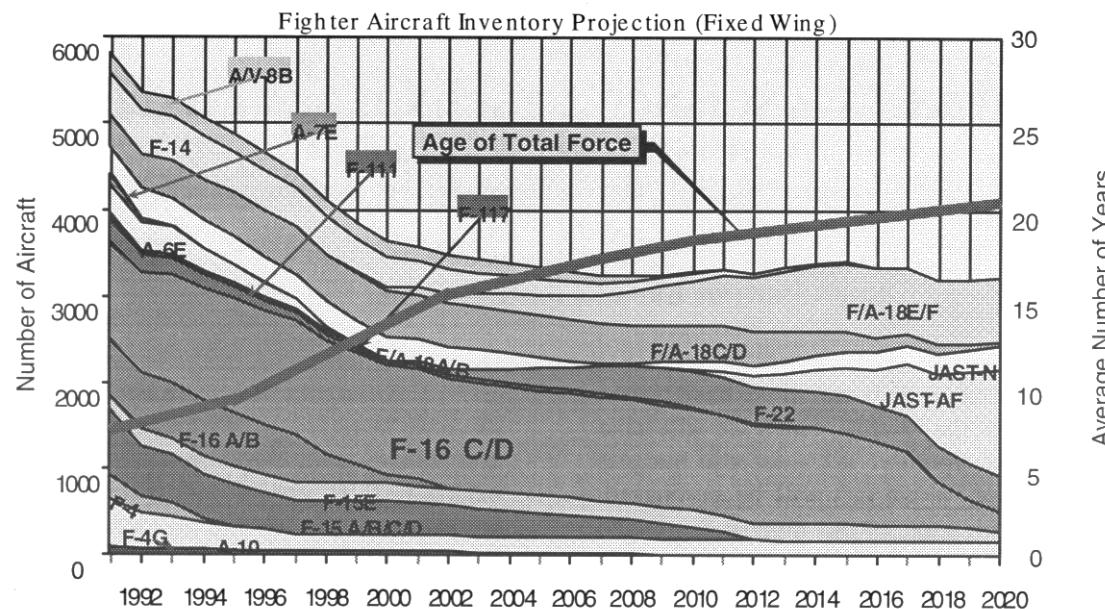


Figure 1 Tactical Fighter Inventory

Clearly, avionics on these aircraft will require modernization during the next decade if these weapon systems are to maintain their readiness status. Future aircraft modernization will focus increased capabilities which require effective communication with active forces, concurrent use of weapons, compatibility with electronic combat systems, up-to-the-minute access to off-board intelligence, an effective means of cooperative targeting, high accuracy bombing with inertial guided weapons, and on-board training for pilots and ground personnel. These capabilities must be inserted affordably and within the cost or schedule constraints imposed by the military budget.

Many technological advances having profound implications for military capabilities are the result of research and development conducted by commercial firms for essentially the commercial markets. Digital products found in personal computers have become the center of focus on commercial technology insertion as they have become much more powerful than many of the computers being used in our weapon systems, and at dramatically lower prices (see Figure 2). The power of the processors gives intelligent weapon systems their capabilities. The combined factors of expanded capabilities and lower costs are clearly compelling reasons to integrate these

commercial technologies into our existing and advanced weapon systems.

Legacy aircraft issues and reasons for upgrade

Recent DoD budgets and the political climate are forcing the United States military aircraft to remain operational well beyond their projected service life. These life cycle extensions have prompted many legacy aircraft upgrade programs. In addition, today's military aircraft must perform an increasingly broad range of missions from tactical/strike to reconnaissance and electronic warfare. These emerging missions encompass a wide range of avionics functions, performance, and cost. Several platforms are in need of upgrades due to the need for increased connectivity, the lack of spares, and directed initiatives that require increased avionics performance/functionality. However, current budgets have been reduced to the point where it is difficult to meet all requirements using the traditional upgrade methods employed during the 1980s and early 1990s.

In the past, piecemeal system enhancements were performed to meet new requirements. These piecemeal retrofits have resulted in point designs that only support a particular problem and platform. As a result,

other platforms could not readily leverage these developments. Many of these upgrades were not applicable to subsequent upgrades of the same subsystem, and as time passed, upgrades were layered upon each other compounding the maintenance problem. Tight coupling of the legacy systems forced the major parts of the avionics system to be recertified, which became a major cost driver for the upgrade. Furthermore, these upgrades must be performed and scheduled into the existing maintenance schedule or the existing block upgrade cycle.

upgrades are currently funded. Some issues are structural in the present procurement and management approaches. Decisions are made on a program by program basis, and we see no trend to decrease the authority of the system program directors. Any changes to aging aircraft avionics will have to make sense to the weapons system program director, while providing capabilities and supportability acceptable to the user command. Introduction of a new supportability concept, such as one- or two-level maintenance, will have to prove itself in the context of the existing supportability plan for the weapon system,

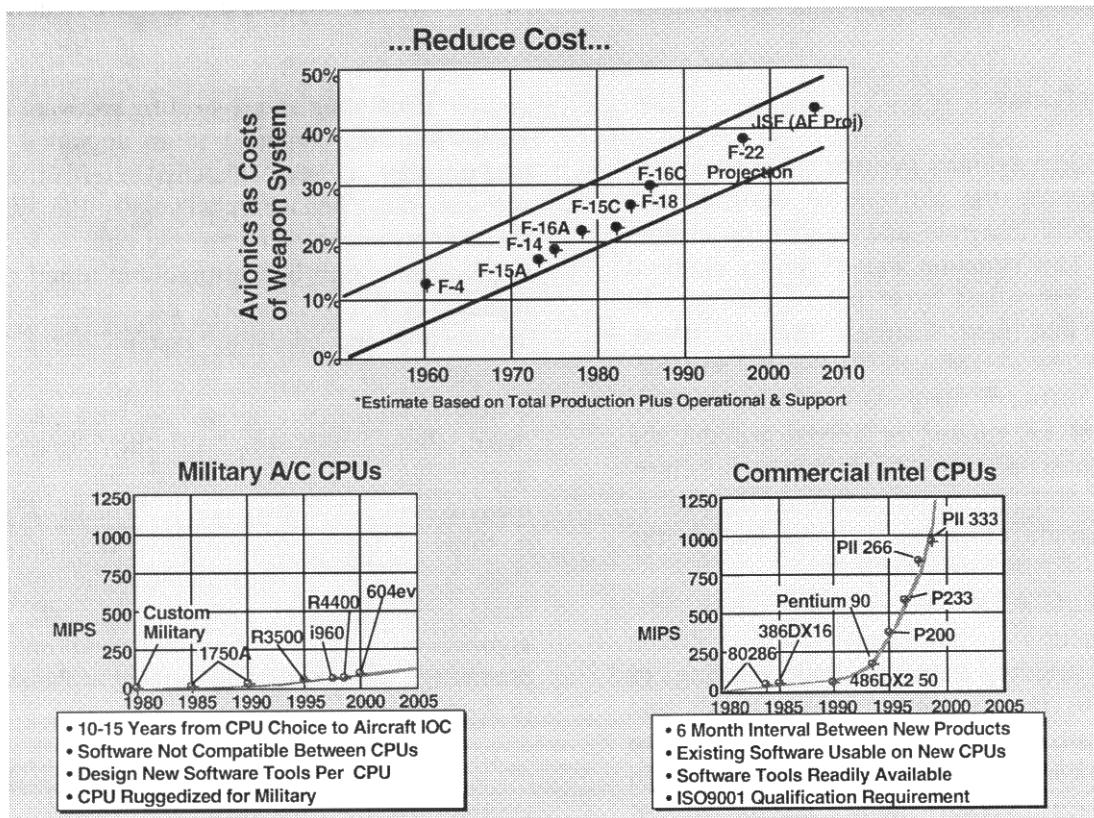


Figure 2 Commercial and Military Processing Technology Comparison

Commercial digital technologies have increased in performance several orders of magnitude over the past decade. Advanced data and signal processing capabilities present opportunities for extending the life expectancy and meeting new mission requirements of our existing military aircraft by inserting new capabilities in weapon system platforms.

Beyond commercial technology insertion and basic logistics problems, there are issues of administration, funding, and organization of how

possibly with a 3- to 5-year pay-back.

Reasons to upgrade

New and enhanced mission requirements. Legacy avionics designs are generally based on technology that is at least 15 years old and at least 2-3 orders of magnitude slower than commercial equipment available today. Thus, legacy designs do not have sufficient capacity to meet requirements demanded by the new and enhanced missions, even if significant spare and growth room is built-in.

Increased reliability and better fault isolation.

Upgrades using modern technology allow for increased reliability because fewer parts and fewer interfaces are required to provide the same functionality, thus providing Life Cycle Cost (LCC) savings potential. This can also lead to better fault isolation because fewer can not duplicate (CND) anomalies are produced.

Component obsolescence. The military qualified components used on legacy systems are becoming obsolete at an increasing rate. This provides an opportunity to upgrade the system using COTS parts and the potential for substantial LCC savings.

Avionics Software

Modern avionics systems are expensive to upgrade because the software is designed and built to tightly integrate and operate on custom hardware. This situation was driven by the need to squeeze every last cycle out of hardware to implement that next feature. Simply stated, current avionics software is not portable. However, software upgrade and maintenance costs cannot be studied or solved without the defining context of a specific system architecture. Given this situation, major software cost drivers for the upgrade of legacy systems include the following:

1. Recovery and understanding the original system requirements and design constraints. Lack of understanding of these requirements causes rebuilds and sometimes program restarts due to erroneous assumptions and poor design choices based upon incomplete or inaccurate information.
2. Interfacing with the legacy design philosophy of the avionics system and sub-systems. This applies for both hardware and software (language) issues.
3. Interfacing to legacy system operating system environment to meet both scheduling and communication interfaces. Often times this is not a clean interface due to the cyclic nature of most deployed systems and the hardwiring of addressing on the communications bus (i.e. 1553). Traditional approaches to solve these problems (e.g., sharing memory, bus snooping, etc.) are error prone because the

new system cannot determine the absolute state of the legacy system.

4. Re-certification and Re-validation of the system and/or sub-system as they are incrementally inserted. The cost of validating the software and hardware for safe and correct operation each time functions change will dominate the cost of any upgrade involving functional modification.
5. Fitting major software/hardware builds and insertion into the normal maintenance cycle. The time to develop and insert a new avionics suite (or a major sub-system) exceeds the standard maintenance cycle.

Upgrading and re-hosting software from a legacy avionics computer to an upgraded one is traditionally a very expensive effort. The software must behave substantially the same on the new processor as on the old one, so interface requirements must be captured and tested. Much legacy code is closely dependent on the architecture and operating system of the legacy computer and is often written in an obsolete computer language. These problems conspire to make direct re-hosting of the code nearly impossible. Several approaches, including reengineering of the code or encapsulating portions of legacy code with modern wrappers, show promise by retaining some of the investment made in the original code development. Other approaches, such as translation and emulation retain even more of that investment at the expense of future maintainability.

Standards apply at all levels of system/software. To aid in the classification of standards, the Society of Automotive Engineers has a reference model called the “Generic Open Architecture model” (GOA). This reference model, shows a detailed breakout of key interfaces for systems. Each of the GOA interface areas are opportunities for reducing the level of interface incompatibility within future systems, improving upgrade opportunities, and providing the basis for procurement guides. For example, IEEE POSIX defines interface standards at the GOA level for operating system and operating system services, and has many commercially available implementations. POSIX allows any application software to be portable to another system that uses a POSIX solution for the level of interface.

Widely commercially used standards (e.g. CORBA or POSIX) provide access to a wider base of professionals that have used the standard, and the implementations of that standard have more operational hours—i.e. fewer defects. This commercial appeal also provides better and greater numbers of available development environments. All these factors lead to lower cost for systems that can take advantage of commercial software interface standards.

Software development cost has been the single major cost driver in most avionics improvement programs. An effective incremental upgrade must utilize COTS technologies to shorten the software development cycle and increase productivity. For MIL-STD-1750 Processor-based software development, few development tools are available, especially for legacy programming languages. Furthermore, it is not uncommon for legacy codes, based on older languages such as FORTRAN and JOVIAL, to become difficult or impractical to maintain as familiarity with the target hardware and highly optimized control and application programs and their development tools diminishes when the engineers who designed them move on. For these reasons, software engineers have begun to migrate away from proprietary, application-specific architectures towards well-defined COTS open systems standards and technologies whenever possible. (See Figure 3) This involves using

system environments like DOS, UNIX, VTRX® (MicroTech), VxWorks® (WindRiver); and interfacing to software Application Program Interfaces (API) like Windows® (MicroSoft) or TCP/IP communication protocols.

Group A modification

Group A modifications involve permanent modifications to flight equipment on the aircraft. Avionics upgrades usually require major Group A modifications at significant cost. Studies conducted by the Modular Avionics System Architecture (MASA) Program on the F-16 fleet indicate that at least three “black box” systems must be integrated into one before modular upgrades can be affordably integrated into a single black box replacement. During this study effort, conducted in 1990 for the F-16, it was concluded that the cost of additional Group A wiring changes could be justified with out-year life cycle cost savings. But as effective service life decreases, the life cycle cost savings are amortized across fewer years. In general, the cross-over point for high rates of aircraft modernization is approximately 10 years of remaining service life. For low rate or low number modernization efforts, Group A development cost is assumed to be the dominate cost inhibitor. In various studies conducted, over twelve change proposals incorporating Group B equipment were altered or

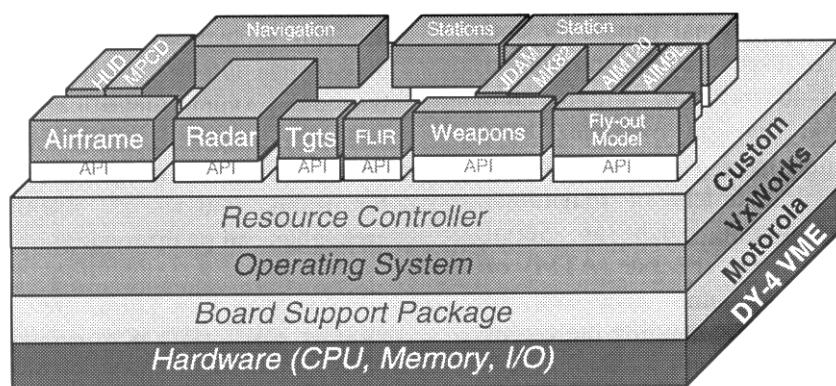


Figure 3 Open Software Architecture

structured, well defined languages like C and C++; employing object-oriented programming methods; working within familiar, broadly supported, hardware independent operating

reduced in scope to accommodate limitations of the Group A changes. One significant program was canceled due to estimations for F-16 Group A cost impacts. The results of this analysis

provided an important assumption for aircraft modernization. If future upgrades are small and incrementally funded, the typical upgrade may be projected as a black box one-for-one replacement, referred to herein as in-place upgrade. As Group B black boxes or LRUs are upgraded, the physical wiring, cooling, power, and mechanical interfaces will largely be left unchanged for the remaining service life.

As part of avionics upgrades, modifications to existing aircraft electrical and mechanical configuration items may be required in order to interface with the new elements. These include power, cooling, wiring harnesses, mounting fixtures, cables, connectors, etc. The modifications can be extensive and time consuming when redesigning of existing Line Replaceable Units (LRU), replacing and adding new wiring harnesses, and on aircraft system integration and testing are involved.

A systematic approach that allows insertion of improvements utilizing wiring, installation, and other existing aircraft resources will drastically reduce Group A modification costs. The practice of using field installable modification kits will further reduce time-to-field and unscheduled aircraft down-time.

In addition to electrical and mechanical changes, a major consideration to upgrading legacy avionics is their network architectures. MIL-STD-1553B Multiplexed Bus structure has been the standard for avionics multiplexed communication implementations. While the demand for faster computational capability and data communication of high volume, intermixed data types continues to increase, bridging the MIL-STD-1553B time division multiplex bus-based avionics architecture (1 Mbits/sec.) into one that utilizes several orders of magnitude higher network throughput, such as Serial Express, Asynchronous Transfer Mode (ATM), or Gigabit Ethernet, has become necessary to avionics modernization. By expanding the use of existing MIL-STD-1553B cabling in conjunction with a scalable protocol implementation, Group A modification cost can be minimized.

Changes to the cooling infrastructure are expensive and usually require changes to the aircraft structure. Thus cooling air found in most legacy aircraft is the heat removal medium. Edge

conduction modules have a higher thermal dissipation capability than convection cooled modules, plus they have better vibration and shock performance due to their central heat sink. It may be possible to use liquid edge conduction cooling with heat exchangers to the cooling air in cases where size restrictions and the resultant thermal density of the modules require greater thermal dissipation capability than cooling air provides. In such cases, the liquid cooling infrastructure (circulation and heat exchangers) may have to be self-contained in the avionics upgrade. However, a prototype avionics liquid cooling system for currently fielded aircraft and for the retrofit of state-of-the-art avionics has been demonstrated. The electrically driven vapor-cycle heat pump was integrated into the F-16's current cooling system and increased the cooling capacity by 5KW thus enabling the aircraft to be incrementally updated well into the next millennium with advanced, reliable avionics at low cost.

Avionics Maintenance

Typical Operations and Support (O&S) costs run approximately 70% of the total Life Cycle Cost (LCC). Cost distribution is spread over Support Equipment, Depot Maintenance Equipment, Personnel, Initial Training, Initial Technical Data and Initial Spares. To reduce O&S costs, avionics systems must be highly reliable and easy to operate. Timely repair or replacements will minimize Mean Time To Repair (MTTR) and increase Mean Flight Hours Between Unscheduled Maintenance (MFHBUM). Legacy aircraft avionics upgrades must introduce innovations such that system and subsystems level diagnostics can be performed to provide accurate, thorough problem identification and isolation. In addition, periodic preventative maintenance can be automated and prognostic data made available for trend analysis. Hence, non-repeatable anomalies can be eliminated.

“Can Not Duplicate” (CND) faults and “Retest OK” (RTOK) problems account for a large percentage of the maintenance events on legacy aircraft. Better diagnostics will help reduce these problems, but some of the solution must come from the design of the system. Reduction in the number and complexity of interfaces will increase reliability and reduce CNDs. Though CNDs should occur less frequently in a reliable COTS-based system, when they occur, a method

must be provided to save the system state in order to duplicate the situation and attempt to diagnose the cause of the CND in a laboratory.

Mainstream commercial processors and systems do not normally have much diagnostic capability other than at system start time. Some systems are capable of isolating memory errors to a specific failed memory module and some operating systems can probe devices to determine if they are operating normally, but other than that, very little isolation is available. More expensive computers designed for high availability, such as servers and supercomputers, have some built in fault detection and isolation along with redundant hardware to help tolerate the faults. Finally, error correcting/detecting memory controllers can be used to help correct for and diagnose failed memory components. These techniques and others need to be investigated to determine how commercial technology-based avionics upgrades can diagnose and recover from failures and aid maintainers in determining where the failure occurred.

Distributed avionics upgrades require system management software to work with the health monitoring software on each processor to maintain a consistent and correct view of the health status of the system. When failures occur, the system manager must activate the programs impacted by the failures on other processing resources. In addition, enough system state must be available to these newly-activated programs so that they can begin operation rapidly and not cause any mission-critical functions to fail for longer than a specified time. An overall system health maintenance and reconfiguration approach must be integral to the upgrade architecture and its associated system software.

Critical to legacy aircraft process evolution is the support strategy and maintenance improvements introduced to legacy aircraft weapon systems. The definition of this support strategy is key to establishing lower operational costs when service life decreases as a function of time remaining in service life. In addition, interfacing the weapon system to the global network is a major goal for future product support systems in order to establish a complete network resource to determine combat status of individual weapon systems for local theater commanders during weapon system deployment. The

technology provides users with automated pilot debrief, advanced diagnostics and electronic technical manual data presentation capabilities. The new technology is being used for current legacy aircraft as well as future programs.

The Integrated Maintenance Information Systems (IMIS) technology is a multi-level secure, integrated, distributed task and decision support system that automates numerous processes to improve pilot debrief and maintenance performance functions. IMIS reduces management, supervisory and technical overhead by providing:

- 1) Real-time access and interface with customer Logistics Data Systems as well as other Department of Defense external systems,
- 2) Aircraft advanced diagnostics capability, and
- 3) Display of Interactive Electronic Technical Manuals (IETMs).

The system reduces the hours needed to service, troubleshoot, and repair all aircraft systems and assists in the achievement of the maintenance mission goals of high sortie rates, to minimize aircraft down-time, and to provide maintenance with minimum support resources. Obtaining accurate data of aircraft availability and status is critical to mission planning and repair operations. Achieving timely and accurate data from field operations is an important aspect to understanding when and if weapon systems require modifications to be effective in their missions. As commercial technology is inserted into legacy aircraft systems, it becomes crucial to monitor the field operations of these equipments, as critical built-in-test and self-test hardware and software is generally not built into commercial grade components and systems. Hence the logistic system must be modified to accommodate and control the infusion of technology into the fleet of aircraft and provide a means for technology refresh, as shown in Figure 4.

Aircraft data is received through a data transfer cartridge (DTC) or via an aircraft maintainer vehicle interface (MVI). Although capable of operating as a stand-alone system, the IMIS interface can communicate logistics data through a network connection to external systems. IMIS consists of a networked computer system that

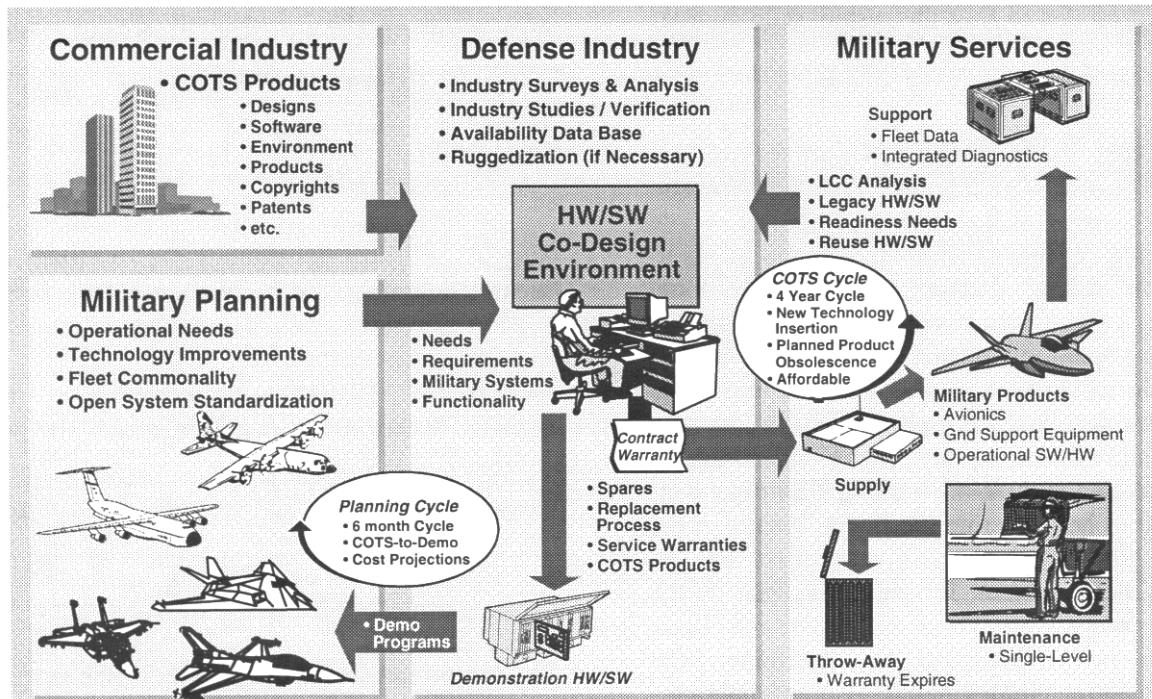


Figure 4 Logistic Support is a Vital Aspect of Technology Refresh

communicates with Maintenance Information Workstations (MIW) or Portable Maintenance Aids (PMA). The system components can be transported, allowing use during deployment. Current IMIS development efforts add IMIS capability to provide the system user with real-time access and interface with customer Logistics Data Systems as well as other DoD external systems.

Anticipated benefits of the IMIS system include:

- 50% reduction in pilot debrief time
- 30% improvement in troubleshooting accuracy
- 30% reduction in Re-Test OK (RTOK) rates
- 40% reduction in the time to repair a malfunction or perform maintenance actions resulting in:
- Improved weapon system availability and readiness
- Reduced aircraft down-time
- Improved aircraft sortie rates

- Reduced weapon system life cycle cost
- Improved maintenance data collection

Currently, IMIS is in F-16 field service evaluation and supporting the F-22 Raptor aircraft. Further evaluation is required to potentially add a commercial interface standard such as Ethernet to allow direct access to aircraft weapon systems, either wired, spread spectrum RF, or high speed IR links are being defined as an open systems approach to field service and support. Another element of the IMIS system is the integration of Interactive Electronic Technical Manuals. The Interactive Electronic Technical Manual system allows the user to locate required information faster and more easily than the current paper technical manuals. IETMs are easier to understand, more specifically matched to the system configuration under diagnosis, and are available in a form that requires much less physical storage space than paper. Interactive troubleshooting procedures have been added using the intelligent features of the IETM display device.

Application of COTS products to upgrades

The use of commercial technology-based components is becoming more and more essential

for avionics upgrades because of the lack of availability of military-qualified parts. In addition, commercial components are normally significantly less expensive and have higher capability than similar military parts. While commercial components are not designed for direct application to military avionics applications, it is expected that methods can be employed that will allow them to operate reliably and cost effectively within avionics environments. The use of COTS components presents other issues that need to be resolved in order to provide a smooth transition to COTS-based avionics, and recoup the cost savings that would make the transition worthwhile.

Due to the differences in the environments of the various legacy platforms, and the resultant variations in requirements, a procurement strategy must be developed that can maximize the benefits of applying COTS to the various platforms. In addition, aircraft will be upgraded in blocks, but the COTS technology used will most likely be obsolete before the block upgrade is complete, so there needs to be a configuration management scheme and continuous upgrade process in place that allows inexpensive insertion of newer components with low-cost revalidation. Methods for certifying the aircraft fleet with the various versions of the block upgrades must be defined. Integration of the COTS-based avionics with the legacy avionics need to be addressed; examples of integration issues include, but are not limited to, interfaces, EMI, and architecture changes. Other areas, such as support equipment and training, will also require attention in order to minimize the cost impacts of the ever-changing COTS technology; this would include such areas as documentation changes as well.

Due to the fact that all aircraft in a fleet are not upgraded at once, some of them will have different hardware configurations than others. Various methods exist for configuring the avionics hardware, both at installation and at maintenance time. One possible technique involves having the avionics system perform a plug-and-play type operation of determining the hardware components it has and loading the appropriate software from a mass storage device.

Another aspect of configuration control is the logistics aspects of the hardware components.

It is possible to configure the aircraft in terms of functionality as opposed to part numbers. Hardware components are made to be Form, Fit, Function and Interface (F³I) identical. The components are then tracked according to their F³I classification and not their part number. Aircraft can then be made to be identical as far as configuration control, and the logistics can be improved due to a reduction in "part numbers" (F³I numbers). Of course, each component will have to meet some minimum requirements and testing to achieve its F³I classification.

Mil-Std-490 has been canceled and replaced with a handbook. Government / Industry Data Exchange Process (GIDEP) interaction with just commercial components needs to be fleshed out along with RADC traceability and controls. Mil Std 965, Parts control program, being canceled has to be dealt with philosophically. Is this just a subcontractor issue? Are we partners?

The current trend of higher clock speeds and shorter rise and fall times for digital circuits, plus the small size and increased sensitivity of analog circuitry, presents an opportunity for Electro-Magnetic Interference (EMI) when COTS components are used in avionics applications.

The increased functionality that can be fit onto a module presents the potential for a high density of circuitry in close proximity. Precautions must be taken to prevent analog and digital circuitry from interfering with each other. Moreover, the newer, higher-speed circuitry may present EMI levels that exceed the levels that legacy avionics were designed for. Thus, precautions must be taken to prevent the avionics upgrade and the legacy avionics from interfering with each other.

Because of the rapid advances in commercial computer technology, COTS-based processors should be the most cost-effective part of an avionics upgrade. It is reasonable, therefore, to examine which traditionally analog avionics functions can be handled inexpensively by these processors. Digital radios are becoming available in the commercial arena, so it makes sense to push digital technology as far as reasonable in avionics upgrades. Digitizing at the aperture could eliminate costly waveguides and high-maintenance analog hardware, replacing it with digital networks and signal processing. This

approach to upgrades may reduce cost, weight, and volume, while enhancing maintainability.

Quality Assurance - Quality assurance (QA) objectives of military systems must be maintained when applying commercial product QA standards established by the International Standard Organization (ISO). It has been demonstrated that streamlined COTS processes and procedures can reduce Life Cycle Costs and also meet the needs of military applications when applied appropriately.

Operational Requirements - Applicable thermal, random vibration, pressure/altitude, electromagnetic and other military operational requirements must continue to be satisfied when COTS elements are introduced in weapon systems. A number of COTS products are presently qualified for many legacy aircraft's avionics operating environments. The associated COTS suppliers have implemented engineering options for meeting different operational requirements.

A technology "bridge" must be continuously maintained into the design of incremental upgrades, so that parts obsolescence and low cost COTS insertion can become facts of life to legacy avionics systems.

Obsolescence and Rapid Design Changes

The life cycle of microcircuit technology is getting shorter and shorter. In the 60's the market availability life expectancy was 20-25 years. The market availability life expectancy of microcircuits from the 70's was reduced to 15-20 years. This was further reduced in the 80's to 10-15 years. Today the market availability life expectancy of new microcircuits is 7-10 years and this includes introduction and phase-out. In reality, for some of the high-performance microcircuits like processors and memories, the market availability life expectancy is less than 5 years. This presents a problem for military systems, which are generally designed to operate for a life span of 20 to 30 years.

It is not uncommon for military avionics to be obsolete before it is deployed. Many modern systems, including the F-22 and C-17, have undergone major avionics upgrades before achieving operational status. Between the changes

in requirements due to the global and national political climates and the rapid evolution of technologies applicable to avionics, the custom architectures created to solve the original problems become rapidly outdated and very expensive to change.

Some of the major cost factors for obsolescence and design changes are described below:

- The current rate of technology turnover in the digital circuit market is staggering. The mean time to obsolescence for all digital parts is around 24 months, and processors are becoming obsolete in less than 18 months. For military avionics, this situation is compounded by the fact that there are a decreasing number of silicon foundries producing military grade parts, and those foundries are producing fewer military parts. It is simply not cost effective for these companies to tie up valuable production capability for a minority market segment. Military parts do not drive the market for digital circuits, and the military supplier must learn to use commercial digital parts to deliver future avionics systems and avionics system upgrades.

Digital components are not the only system components that are affected by technology change. Software interface standards, bus standards, protocol standards, network standards, and others all change at different rates. Each of these system components are cost risk factors for new or upgrade avionics systems.

- The non-recurring engineering (NRE) required to capture the existing system interfacing requirements is non-trivial. The system documentation does not contain the wisdom of the original designers/implementors, and many times these people are no longer available. Further, the documents do not reflect the current state of the system -- or are missing the fine details to make the system really work. Accurate specifications of the system interfacing and function are required to upgrade avionics systems.
- Beyond the problems of understanding the system interfaces and function, the upgrade avionics system must pass several re-certification levels. Flight qualification, system acceptance, and a number of other tests are required before the upgraded system can be deployed. The cost of re-certification

of avionics systems *can be a major cost driver*.

- One of the major operations and support cost is the logistics to track and store spare parts for aircraft systems. If lifetime buys are considered, several cost factors must be contemplated. First, the sunk cost of the lifetime buy. Second, the additional logistics cost for tracking and storing more parts. Third, the cost risk of those parts becoming obsolete before the parts are used. Finally, the cost of monitoring the obsolescence status of parts within a design. All these cost factors must be considered for the lifetime buy option.

Parts obsolescence is also an issue for military components but not nearly as severe as it is in the highly-competitive COTS environment (see Figure 5). Provisions for the management of obsolescence need to be in place from the beginning of the design process. Obsolescence considerations must be part of any trade-off studies performed in order to select candidate COTS technologies for use in a military system with an extended operational life. One way to manage obsolescence involves drastic changes in the way the military acquires systems. It entails having a number of short development and procurement cycles throughout the life of the

system, where upgrades are incorporated into the system. These short development cycles would include the required testing to recertify the integrity of the system. If done right, this approach can eliminate the need for many of the logistic support requirements. If the equipment is highly reliable and low cost, only a small number of spare parts will be required to sustain the system in operation. Upgrades could be available before any failures occur, eliminating the requirements for a complex logistics support structure.

A planned obsolescence approach is required for systems using technologies with a high rate of change. One approach to address this technology change is to develop an overall system architecture that uses well defined (and widely used interfaces) for those items that are subject to rapid obsolescence. Widely used interfaces will tend to have support strategies to get to the next standard (e.g., interface chips that bridge the old interface to the new one). Less popular (or even custom) solutions can be used for system architecture components that do not change rapidly (i.e., antennas or radio frequency amplifiers).

From this baseline upgrade open

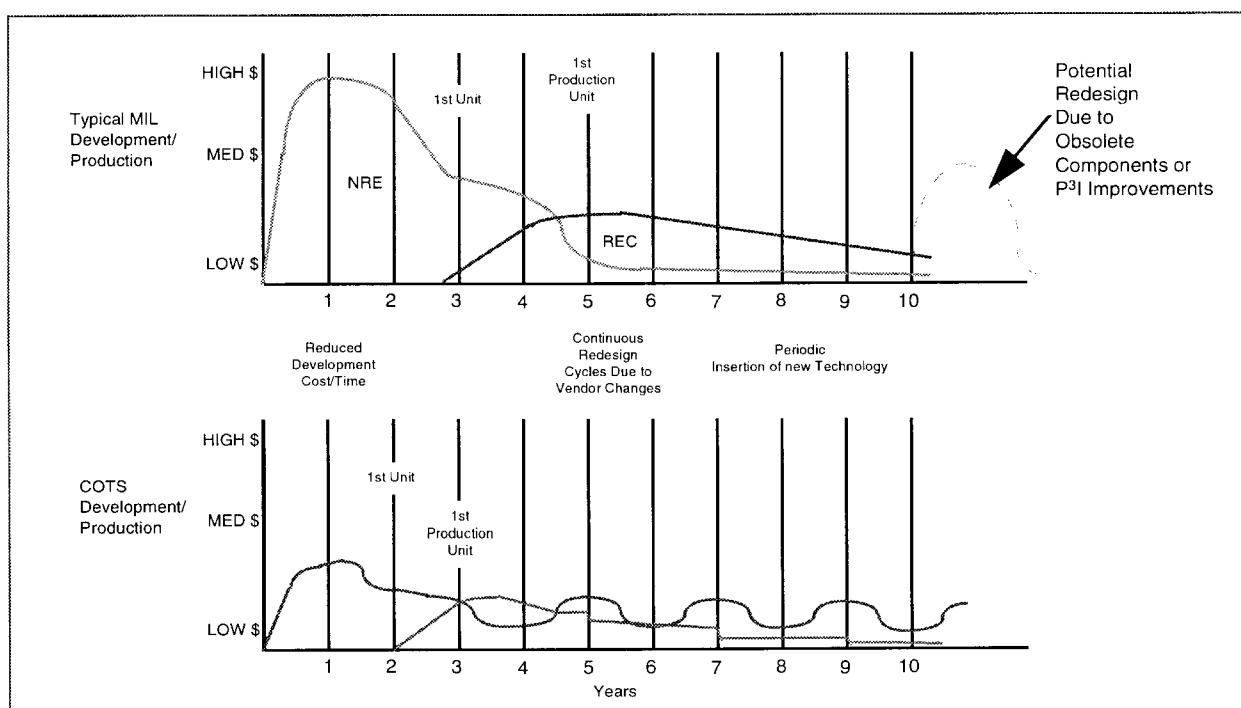


Figure 5 Contrasting Program Costs

architecture, portions of the architecture can be inserted incrementally until the system achieves a fully upgraded state. Then as components of that architecture become obsolete, they can be easily replaced with newer generation parts. Part of this approach is the establishment of an obsolescence-management contractor-based sub-IPT early in the program. The IPT would be in charge of:

- Performing detailed obsolescence surveys of potential suppliers
- Maintaining current survey results on file
- Identifying single point-of-contact for obsolescence notification in each of the sub-contractors involved in the program
- Monitoring and maintaining life-cycle ratings for all the active technologies being used in the program
- Rating of technologies longevity by families
- Performing program baseline reviews
- Establishing and maintaining a master parts file repository
- Recording and maintaining a lessons-learned file.

A design capture process using “model-in-the-loop” techniques also helps solve problems with obsolescence of commercial technology-based upgrades. The design capture process results in a detailed executable specification that can be verified using system acceptance tests and other real-world inputs. This executable specification can then be implemented in various ways, including a VHDL-derived replacement part, software emulation, or a modern-technology software simulation. When the first implementation becomes obsolete, due to unavailable parts or for other reasons, a new instantiation of the system can be derived from the executable specification at a substantially reduced cost. This new replacement may or may not use the same implementation choice as the first upgrade, but since the executable specification already exists, little or no time needs to be spent deriving requirements or defining test procedures.

By using executable specifications (e.g., VHDL) to describe digital components, replacement parts that meet the functional

specifications can be achieved. Commercial industry routinely uses VHDL to fabricate or program gate arrays and other programmable logic families (e.g., FPGAs, PALs, GALs, etc.) via commercially available synthesis tools. While not perfect, this process can be used to describe large levels of functionality (e.g., processor and I/O boards). Other interfaces (e.g., physical, electrical, ...) can be achieved through the use of custom packages or adapters. This approach is possible because of the rapid size and form-factor reductions being seen in the programmable logic arena.

Open System Implementation

The primary system implications associated with the use of COTS/Best Commercial Practices occur at the avionics architecture level. A major concern is that, taken to the extreme, reduced contractor guidance could result in a “hodgepodge” of custom boxes, modules, display, etc. which will create an integration and maintenance nightmare, destroy competitive procurements (competitors cannot determine how to build compatible hardware and software) and make common, interoperable avionics impossible. Obviously, measures must be taken to avoid this situation.

One measure that has the potential of reducing some of these problems is the use of an Open System Architecture (OSA) for future and current systems. Definitive guidelines of how OSAs will be employed are currently under review by a DoD task force. The OSA approach is aimed at ensuring maximum competition and common avionics, through a readily available set of system specifications that provide adequate information to build interface-compatible hardware and software.

The key is to define an OSA that will permit an incremental upgrade strategy utilizing modular, scaleable components which can be inserted into the diverse set of fielded legacy architectures. This architecture will provide a framework that can be installed in affordable increments into legacy aircraft while retaining coherent system functionality. The ability to incrementally upgrade a system is important, not only due to budget realities, but also because it is typically impractical to complete a major system insertion/replacement within a single normal

maintenance cycle. Aircraft downtime for kit insertion, functional validation and training must be taken into consideration for reducing time-to-field in both engineering and manufacturing development and production phases of an upgrade.

The incremental OSA approach gradually migrates the legacy system to a new coherent, easier-to-upgrade architecture that avoids the current patching problem. These incremental upgrades can be accomplished in less time and at lower cost than would otherwise be imposed by traditional upgrade methods. Closely related to the installation problems (Group A) described earlier are functional interface issues. If incremental and continuous upgrades are to be successful, well defined functional interfaces must be established between the portion that is being upgraded and the portion that is not (see Figure 6). The purpose of these interfaces is to establish boundaries so that the upgraded portion may not affect the

difficult to apply to existing systems on an incremental basis. Open systems, properly executed, promote competition and provide value to the customer at lower cost. Distributed open systems are an alternative that bears study, along with domain-specific architectures for displays and display processing, data and satellite communication links, and mission processing upgrades. Specific aspects of an upgrade architecture, such as software re-hosting, would be applied system wide. The issue is the cost of changes to the existing system, including test and maintenance equipment, procedures, and documentation, versus the benefit of a new integrated architecture.

Open interfaces that satisfy widely-used standards are essential to implement incremental upgrades. One upgrade can then benefit from the investment and knowledge base of the previous one, such that we do not waste money solving the same system problem over and over again.

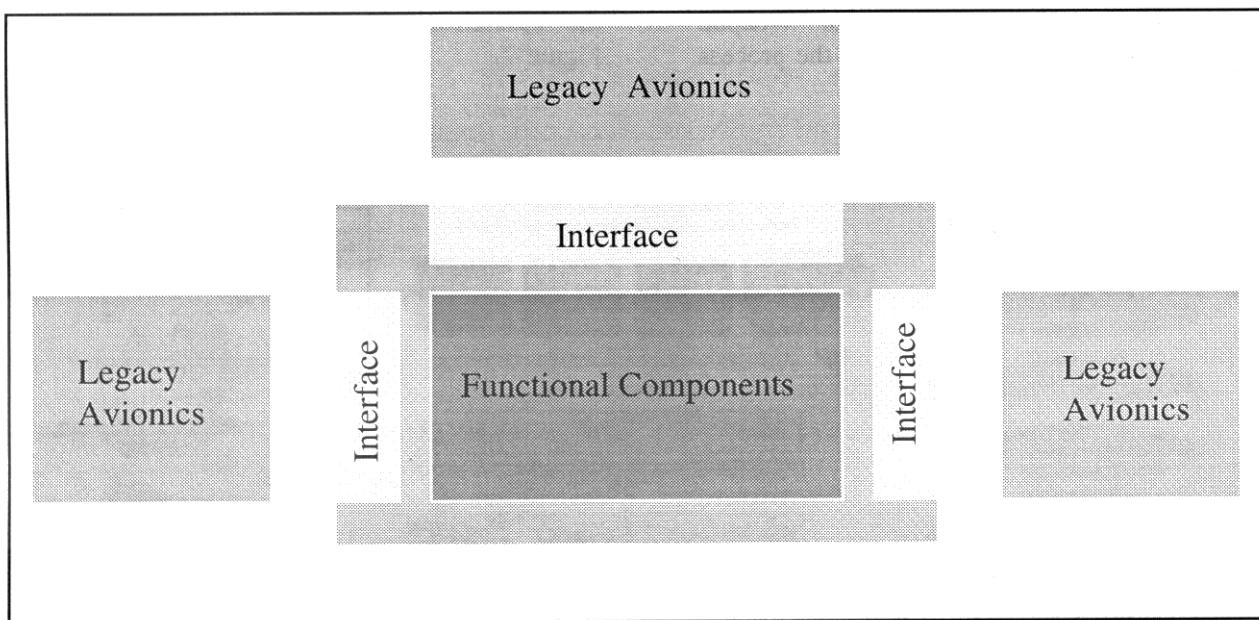


Figure 6 Upgraded Avionics Integrated Through Well Defined Open Interfaces

functionality of the portion that is not being modified. If this is not done, a “ripple” effect is likely to occur where changes affect parts of the system that don’t need to be changed. Not only does this unnecessarily increase the life-cycle cost, it also leads to the unwanted piecemeal designs described above. While new weapon system developments are emphasizing open integrated architectures, these concepts may be

Previous attempts at common module and open interface programs have failed to provide for the future insertion of rapidly evolving digital technology, or have selected interfaces whose usage is limited almost entirely to military systems.

Avionics upgrades generally involve changes to both hardware and software, usually to

accommodate new functional or performance requirements, but occasionally to reduce cost of ownership. We expect that affordable upgrades will become even more of a challenge as the number of software lines of code in our fielded systems increases dramatically during the next decade.

Piecemeal, single-platform upgrades are here to stay as long as the acquisition process is governed by individual program offices and funding horizons are short term. Incremental upgrades are desirable over piecemeal upgrades given the reality of the current budgeting process. Present approaches do not institutionalize the relevant methodologies and architectural concepts, but rather force them to be relearned each time by individual contractor teams. The result is a much longer and more expensive upgrade activity, every time.

Summary

Industry has embarked on a product-line implementation strategy. The major Weapon System/Aircraft Manufacturers are in the process

of defining an open system avionics architecture based on commercial technologies and processes which will be applied across the product lines which they manufacture. Much of this activity is focused on aging aircraft avionics issues to extend the service life of current aircraft coupled with declining military budgets and a dwindling supplier base which challenges the effectiveness of today's front line weapon systems. This approach has been developed by working with numerous government agencies and across the services as well as internal company funding to produce a focused strategy for maintaining the effectiveness of legacy systems while charting a future business strategy similar to large commercially oriented corporations.

Effective use of advanced technologies derived through Science and Technology programs can only be accomplished through a continuous Demonstration and Evaluation Program. This program must be a government/industry partnership aimed at the transition of needed advanced technologies from the research environment to the warfighter. (See Figure 7)

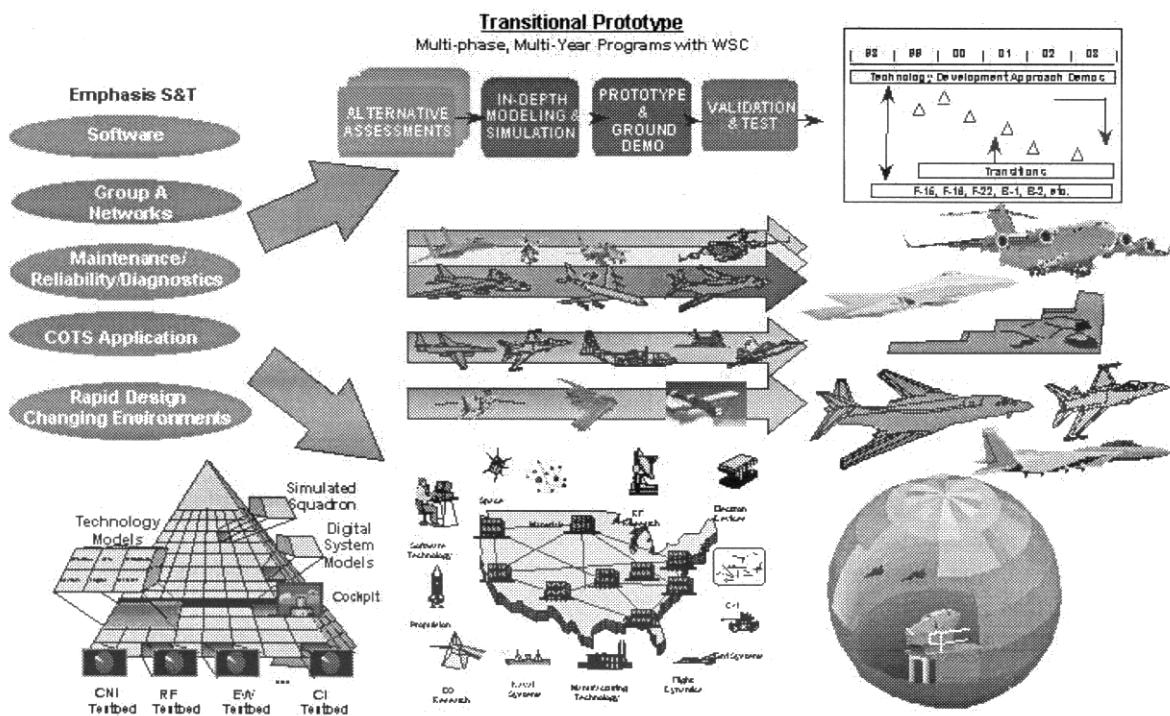


Figure 7 S&T Transition Methodology

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Structural Integrity and Aging-Related Issues for Helicopters

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INTRODUCTION

The question of the structural integrity of aging aircraft became an issue of grave concern when an Aloha Airlines Boeing 737 suffered major structural damage in April 1988 while in flight. Since then the airworthiness issue of aging aircraft has been the concern of manufacturers, and civilian and military operators alike. The issues for civilian and military operators are structural integrity and reduced ownership cost when the service life is extended. The military have the additional task of maintaining preparedness with improved availability and enhancing the performance of aircraft designed for now-obsolete missions to meet new mission requirements. Aging, therefore, does not mean "old" in terms of the number of calendar days, but the cumulative effect of technical obsolescence, changing requirements, quality of maintenance and the nature of operation (i.e., load and environment).

The issues of structural integrity for rotary-wing aircraft are somewhat different from those of fixed-wing aircraft. In helicopters, the dynamic rotor components are safe-life designs and are replaced at the end of their service lives. Thus, airworthiness concerns of structural integrity for helicopters are limited but still pose great challenges in adjusting to changing missions. Structural integrity issues for helicopters are in the airframe, equipment and avionics, and retention hardware for non-airframe related structures. Most of the rotary-wing aircraft in the U.S. Army's inventory are several decades old, and are required to continue in service even longer, Table 1 (Ref. 1). They were designed for missions that have changed and with equipment that have been overtaken by technological advances. Thus, the primary issues for aging military helicopters have been to assure structural integrity while enhancing performance with more capable dynamic components and technically advanced equipment.

Table 1 Age of Rotary-Wing Aircraft in the U. S. Army (Ref. 1)

Type	Number of Aircraft	Average Age, Years	Retirement Date
OH-58 A/C	474	35.0	2017
OH-58D	387	12.5	2024
CH-47D	466	20+	2033
AH-1	389	30+	2017
UH-1H/V	1073	29.0	2025
UH-60A	906	17.0	(Not Set)
UH-60L	515	6.0	(Not Set)

The average age of aircraft in the U.S. Navy's inventory show a similar trend for rotary- and fixed-wing aircraft, Fig. 1 (Ref. 2). The average age of helicopters is 19.2 years and continues to climb until 2005, when aggressive procurement of new helicopters will lower their average age. To continue to field these aircraft, their upgrades should be a planned, continuous improvement process in order to dovetail each stage of upgrade and prevent the cost associated with a one-time upgrade. Because of the severe environment in which the U.S. Navy helicopters operate, it is more convenient to replace life-limited parts than to inspect them to damage tolerance or other aging aircraft requirements in order to assure structural integrity under extended service-life procedures, Ref. 3. Damage tolerance for the U.S. Navy is a "band aid – short-term solution," only to "maintain immediate flight safety, eliminate the problem and return to safe-life operation."

To focus on the structural issues of aging helicopters, the causes of worldwide accidents of civilian helicopters in 1999 are examined in Fig. 2, Ref. 4. The majority of the accidents were due to pilot error followed by engine failure or power loss. Six structural failures were the causes for 3.1 percent of the accidents. The six structural failures were: (1) the tailboom separated on a Bell 407, (2) the retaining nut and bolt for the tail rotor shaft failed on a Bell 206L1, (3) "major mechanical failure" on a Hughes 369D, (4) a main rotor tension-torsion strap failed on a BK 117B-1, (5) a HH-43F was seen to "explode and disintegrate in flight," and (6) bolts in the mounts of the 90° gear box failed. The tension-torsion strap is a safe-life design and is replaced at the end of its service life. The failures of the tailboom, the tail rotor shaft retaining system and the mounts of the 90° gear box are structural issues concerning aging helicopters. The age of these aircraft and the precise causes of structural failure are not known.

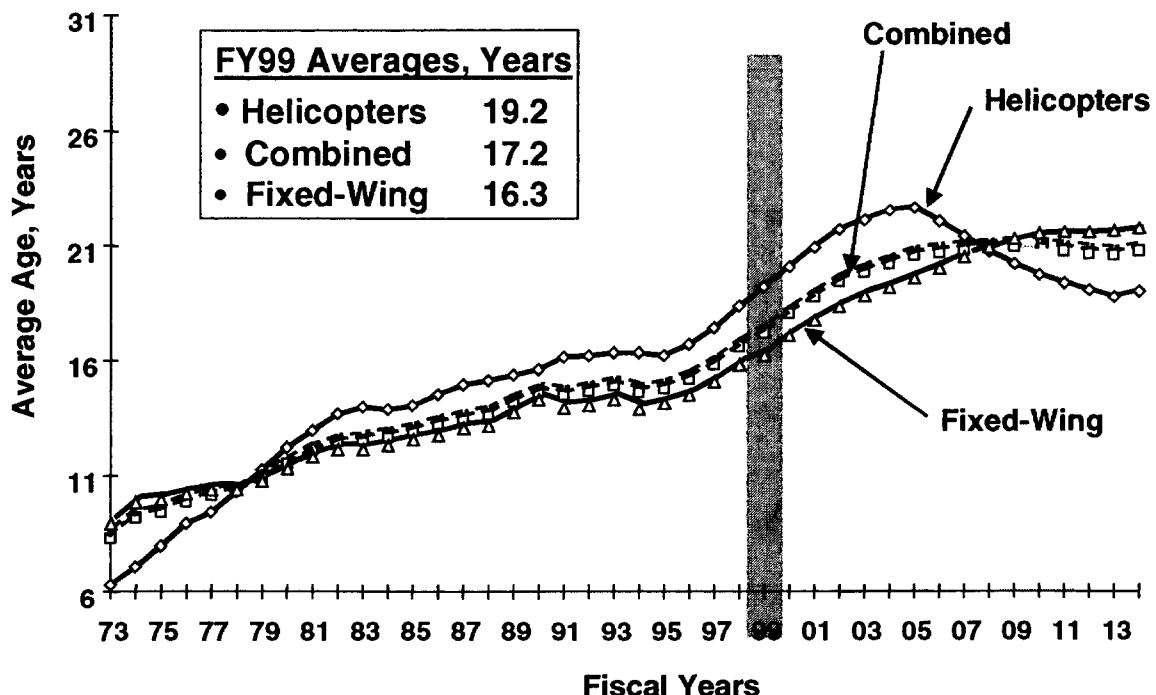


Fig. 1 Average age of U.S. Navy aircraft (Ref. 2)

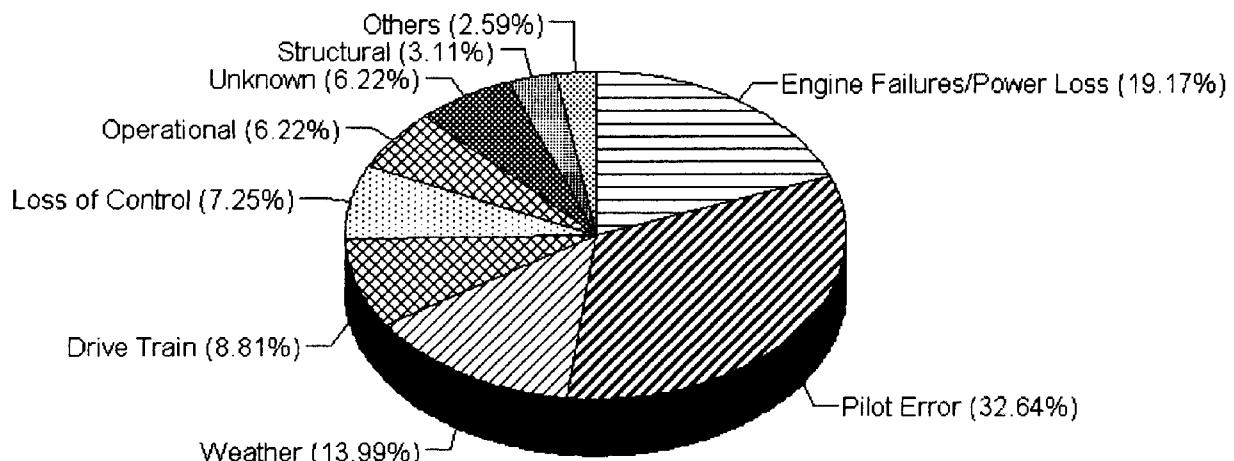


Fig. 2 Causes of worldwide civilian helicopter accidents in 1999 (Ref. 4)

The aging issues among helicopter operators, ordered in terms of the total cost, are engines, dynamic components and drive train, equipment, airframe, and avionics. For medium and heavy helicopters, the breakdown of the acquisition cost and direct maintenance cost (DMC) as percentages of the total cost are shown in Fig. 3, Ref. 5. The total cost is the sum of acquisition cost, DMC and operating cost. The percentages for insurance, fuel and

crew – elements of the operating cost – are also shown in the figure.

Ignoring the cost of labor, the DMC for engines, dynamic components and drive train, avionics and equipment are much higher than that for the airframe. The DMC for airframe is lowest at 10 percent of the acquisition cost, while that of dynamic components is

138 percent. The DMC of dynamic rotor components is high because of replacement cost following their useful service life. The engines and drive train are specialized designs and their aging issues will not be discussed here. Engine and drive train designers are contributing to enhancing reliability through simplified designs and higher power-to-weight ratio systems. Avionics will also not be discussed as avionics designers address cost and reliability through the development of a family of avionics systems, and address continuous upgrades with open operating systems. Equipment includes fuel cells, wiring harnesses and retention hardware of all systems. Wiring systems "age" because of environmental conditions, vibration, operational wear and tear, and improper repair.

This overview presents the structural integrity issues in extending the service lives of dynamic components and airframes of aging helicopters. With the use of composites, the acquisition cost and DMC of dynamic components have reduced greatly over the last two decades. This reduction has been possible because innovative designs of complex geometry can be fabricated more accurately and cost-effectively with composite manufacturing technologies. In addition, significant developments have occurred to make rotor design more efficient, Ref. 6. However, the methodologies for calculating service lives and their optimum applicability are still being discussed. These issues on structural integrity are presented and discussed below. Even though all-composite dynamic components have been in service for several decades, an all-composite airframe is still not a reality. The total composite content in the airframe and rotor of production

helicopters in 1992 was around 25 percent, Fig. 4. Thus, some of the aging issues of fixed-wing aircraft are still applicable to metal airframes of the helicopter though the problems are less severe.

STRUCTURAL INTEGRITY ISSUES FOR HELICOPTERS

In the design of rotorcraft structures, the objective of assuring structural integrity is to reduce to zero the probability of catastrophic failure. Rotorcraft structures can be classified as two distinct types: dynamic components and non-dynamic components of the airframe. Dynamic components are those of the rotor systems and are subjected to high-cycle oscillatory loads. As shown in Fig. 2, loss of structural integrity was the cause of 3.1 percent of the 1999 accidents surveyed. To get a better understanding of the typical causes of, and the frequency of occurrence of these causes in, rotorcraft accidents, an Eurocopter study of accidents over five years on a worldwide, all-missions basis identified 37 accidents per one million flight hours, Ref. 7. Of the 37 accidents, 16 resulted in fatalities. The precise causes of structural failures were identified and were found to be responsible for only 0.3 percent of the accidents, while incorrect maintenance was responsible for 17 percent and engine malfunction for 3.1 percent, Fig. 5. The predominant cause, constituting 77 percent of all accidents, was operational and environmental conditions. The 0.3 percent from structural failures were due to "poor design, non-conformity of components, more severe load spectrum than expected" and non-identified causes of fatigue cracks.

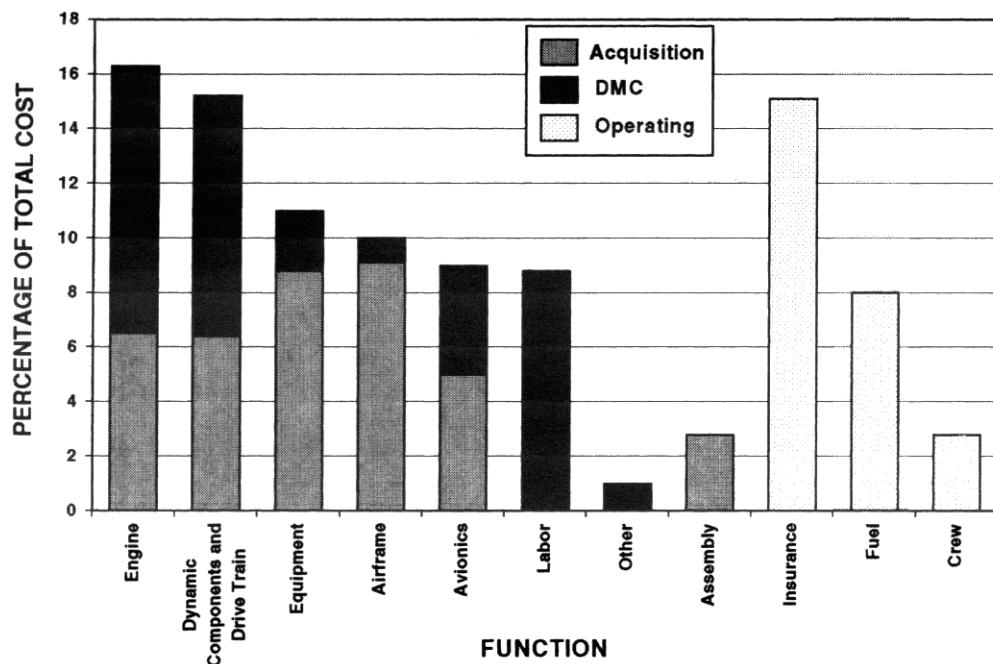


Fig. 3 Breakdown of costs for medium and heavy helicopters (Ref. 5)

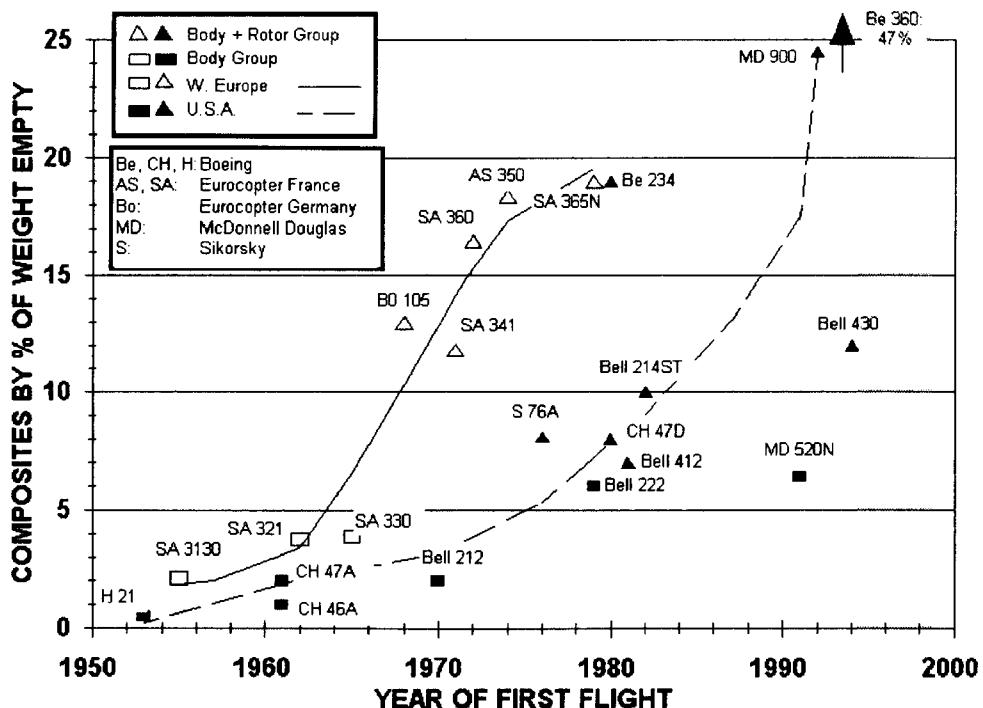


Fig. 4 Application of composites in helicopter to 1992 (Ref. 6.)

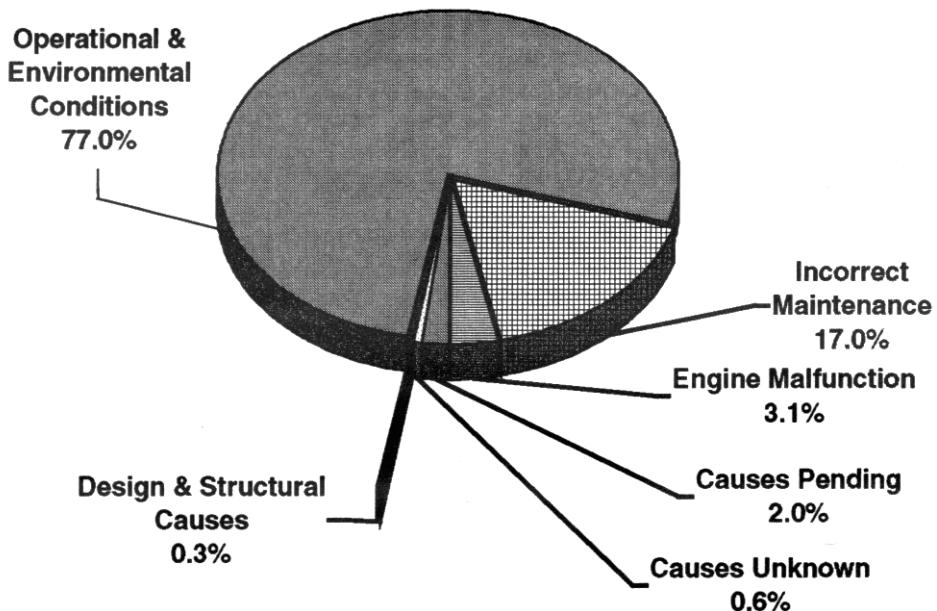


Fig. 5 Causes of 37 rotorcraft accidents per a million flight hours (Ref. 7)

The causes of, or catalysts for, fatigue cracks leading to failures can be assigned to the development, manufacturing and operational processes of rotorcraft.

In development, causes include

- design oversights,
- underestimating loads, and
- analyses oversights.

In manufacturing, causes or catalysts include

- planning oversights
- non-conformal parts, and
- manufacturing defects.

In operation, causes or catalysts include

- usage more severe than expected,
- corrosion,
- improper or inadequate service, and
- service defects.

The defects introduced during manufacturing or in service are not causes but act as catalysts that initiate fatigue cracks, which may then lead to failures. The survey of helicopter accidents in Reference 7 identified catalysts for the cracks, and the frequency with which they occur is shown in Fig. 6.

Rotorcraft manufacturers validate structural integrity and assure, through test and analysis, that the level of baseline integrity will be maintained during operation. The manufacturers also provide guidance on how this integrity can be continued to be assured in service through inspection, and what provisions are available when the integrity is sufficiently degraded where safety of flight will be jeopardized.

Of all the structures fabricated by helicopter manufacturers, dynamic components are the key to the helicopter's performance to stated requirements. Dynamic components also have the highest DMC compared to all other components, Fig. 3. Dynamic components are subjected to large numbers of spectra of oscillating loads and generally fail in fatigue. Several approaches are taken by manufacturers to certify dynamic components. The terminology used to describe these approaches will be first defined, followed by descriptions of, and discussions on, the substantiation methodologies. The aging-related issues and the differences between certifying metallic and composite components are included in the discussions. The merits of the various definitions of the terminology are not discussed and the primacy of any one methodology is not championed.

There are four "traditional" approaches for substantiating the life of components in order to assure safety of flight: safe-life, damage tolerance, fail-safe and flaw tolerance. These four approaches form the basis of the decision process on whether a part be retired and replaced based on accumulated flight hours, or retired based on the damage sustained, or returned to service after repairing

the damage. From practical considerations, components may be substantiated by any one of the approaches or by a combination of several approaches. The definitions of the four traditional approaches are given below.

Definitions

Safe-Life

Safe-Life is defined by the US Federal Aviation Administration in Reference 8 as,

"Safe-Life of a structure is that number of events such as flights, landings or flight hours, during which there is a low probability that the strength will degrade below its design ultimate value due to fatigue cracking."

The safe-life approach assigns a finite life to a component. This definition focuses on the baseline strength and its degradation in operation. The definition also infers, and its application alleviates, the difficulties and cost of inspecting complex rotorcraft structures. The static and fatigue strengths and their progressive losses can be determined through test, or in combination with analysis. The safe-life approach is based on the remote possibility of a crack initiating in a component, and it recommends that the component be retired when accumulated flight hours have completed the assigned finite life or when a crack is detected by currently available means.

Composite components are never as pristine as metal components. In contrast to metals, the influence of detectable damage in composites is difficult to assess because composites often exhibit "cosmetic" damage, which should not be the basis for limiting the structural life or be the reason for redesign. Thus, for a composite part to be retired, the damage must exhibit structurally significant delamination, splintering, matrix cracking and fiber breakage.

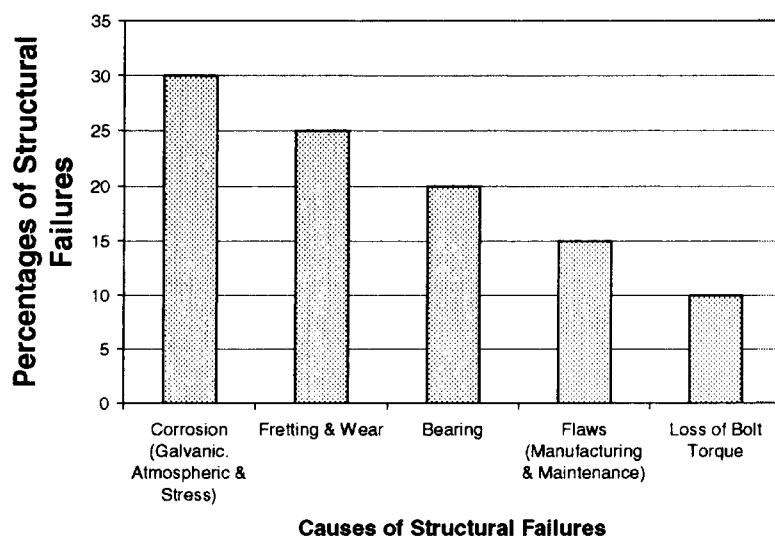


Fig. 6 Frequency of the causes of the structural failures of Fig. 5 (Ref. 7)

Damage Tolerance

Damage tolerance is defined by the U.S. Air Force in Reference 9 as,

"Damage tolerance is the attribute of a structure that permits it to retain its required residual strength for a period of unrepaired usage after the structure has sustained described levels of fatigue, corrosion, and accidental or discrete source damage such as (a) unstable propagation of fatigue cracks, (b) unstable propagation of initial or service-induced damage, and/or (c) impact damage from a discrete source."

The focus in this definition of damage tolerance is to quantify the level of damage that a structure can tolerate and retire or repair it before a catastrophic failure occurs. It assumes that any structure is essentially imperfect as a result of the "inherent material structure, material processing, component design, manufacturing or usage," Ref. 10. In order to quantify the level of tolerable damage, the damage must be assessed, and the rate at which this damage will propagate and the damage level at which the residual strength will fail to react the loads must be calculated. The principles of fracture mechanics are used to evaluate the damage tolerance of a structure, and to calculate the period and level of inspection required to mitigate the risk of failure.

The elements of the procedure for calculating the time for crack initiation and the rate of crack propagation for metals are applicable to composites. The difference, however, is in inspection. Inspection is a subjective process, and assessing "damage" in composites is more than measuring the length of a crack. Therefore, training should maximize on the inspector's experience, the type of structure and material in order that the damage criterion for composites is applied uniformly.

Fail-Safe

Fail-Safe as defined in Reference 9 states

"Fail-safe is that attribute of the structure that permits it to retain its required residual strength for a period of unrepaired usage after the failure or partial failure of a Principal Structural Element (PSE)."

A PSE is an element of the structure whose integrity is essential for maintaining the overall structural integrity of an aircraft. Even though the fail-safe concept states that residual strength is essential to achieve redundancy, the concept does not develop inspection requirements to monitor damage. Fail-safe designs, therefore, provide multiple load paths with redundant structures such that the failure of one load path will safely distribute the applied loads to other load-carrying members.

Flaw Tolerance

The flaw tolerance approach is based on the premise that flaws exist in any structure, and act as catalysts for

initiating cracks. The flaw tolerance approach advises inspection and recommends that a component be retired after the prescribed flight hours are accumulated or when a crack is found in inspection. This approach does not recommend periodic inspections to monitor crack growth or subscribe to any residual strength requirements. However, it does require the maximum tolerable flaw sizes in critical locations be determined based on historical data. The data on the size and the location of flaws are then used to conduct constant-amplitude fatigue tests of flawed specimens. Replacement times are then calculated using the safe-life approach. This approach is also known as "enhanced safe-life," and is "sometimes used in combination with or in place of" flaw tolerance, Ref. 11.

SUBSTANTIATION METHODOLOGIES FOR DYNAMIC COMPONENTS

All components subjected to oscillatory loads above the endurance limit will accumulate damage, which result in cracking and wear, and inevitable failure. The helicopter manufacturers have tended towards two methodologies depending on the material of the component, its design configuration, its load spectrum and how critical is its function in the helicopter's operation. The two methodologies are safe-life and "extended safe-life." "Extended safe-life" is a new terminology, and is used in this context to represent a combination of flaw tolerance and damage tolerance. Included in the extended safe-life approach are features of the fail-safe approach where multiple load path designs are featured in critical structures. The features of the four traditional approaches that constitute the two methodologies are summarized and compared in Table 2. The extended safe-life methodology was accepted as viable by representatives of several U.S. helicopter manufacturers represented on the Fatigue Methodology Committee of the Rotorcraft Advisory Group under the Civil Aviation Council of the Aerospace Industries Association, Ref. 12. The improved substantiation process proposed by Eurocopter, Ref. 7, encompasses elements of the extended safe-life methodology described here.

Safe-Life Methodology

Traditional fatigue tests on full-scale components are conducted to characterize their fatigue behavior in the form of S-N curves. The tests are conducted on as-manufactured parts and are subjected to constant amplitude loads based on measured flight loads. The service life is computed with a high safety factor based on the component fatigue strength. The high safety factor applied to as-manufactured parts has to-date accommodated the strength-reducing flaws in structures with a high degree of reliability as shown in Figures 2 and 5. The safe-life methodology, therefore, assumes that a structure has a finite fatigue life which can be estimated from experimental results and analysis. The finite life, or the safe service life, determines the

Table 2 Summary of Substantiation Methodologies

Feature	Extended Safe-Life			
	Safe-Life	Damage Tolerance	Fail-Safe	Flaw Tolerance
Focus	<ul style="list-style-type: none"> Baseline strength Strength degrades in service 	Level of tolerable damage quantified	<ul style="list-style-type: none"> Redundant or multiple load paths 	<ul style="list-style-type: none"> Maximum flaw size in critical locations
Retirement Criterion	<ul style="list-style-type: none"> Prescribed flight hours accumulated Crack detected 	<ul style="list-style-type: none"> On-Condition, i.e., damage degrades residual strength below acceptable level 	<ul style="list-style-type: none"> Residual strength below acceptable level when a redundant load path fails 	<ul style="list-style-type: none"> Prescribed flight hours accumulated Crack detected
Structure	<ul style="list-style-type: none"> Assumed pristine 	Assumed flawed	<ul style="list-style-type: none"> Assumed pristine 	<ul style="list-style-type: none"> Assumed flawed
Periodic Inspection to Assess Criticality of Crack	<ul style="list-style-type: none"> No 	<ul style="list-style-type: none"> Yes 	<ul style="list-style-type: none"> No 	<ul style="list-style-type: none"> No
Substantiation Methodology	<ul style="list-style-type: none"> Traditional fatigue 	<ul style="list-style-type: none"> Fracture mechanics 	<ul style="list-style-type: none"> Traditional fatigue 	<ul style="list-style-type: none"> Traditional fatigue based on flawed specimens

accumulated flight hours allowed before a part is replaced or retired.

The basic elements of the safe-life methodology for metals are constant-amplitude fatigue tests with accelerated loading to develop the component S-N curve, standard S-N curve shapes, usage spectrum developed from measured flight loads, standard safety reduction factors, and Palmgren-Miner's Rule of linear cumulative fatigue damage. In the case of composites where the S-N shape curve is typically flatter than for metals, load excursions, the cycles to crack initiation and the Palmgren-Miner's Rule can greatly influence the component life calculated from the safe-life methodology.

Composite designs are generally damage tolerant. However, because the S-N shape curves of composites are flatter than of metals, they are sensitive to load excursions. A relatively small increase in load results in a large decrease in the allowable number of cycles on the S-N curve. If this load increase is associated with steady-state flight regimes or with frequently occurring maneuvers, the calculated retirement life based on the safe-life methodology could be dramatically reduced.

The safe-life methodology requires a component be retired when a crack is initiated. In composite structures matrix cracking is typically the damage mechanism that leads to delamination. It is difficult to determine when the fatigue strength of a composite structure has significantly degraded. Since the number of cycles to crack initiation is plotted to estimate the service life, an error in plotting it on a comparatively flat S-N curve can result in a greatly erroneous service life.

Since the S-N curve is flatter in composite structures, damage is generally due to a few high loads in the spectrum, such as ground-air-ground loads. The damage may result in several matrix cracks and delaminations. The number of cycles to failure could be the cumulative effect of their initiation and propagation. Since the load representing the cumulative effect is difficult to quantify, Palmgren-Miner's rule must be cautiously applied.

Extended Safe-Life

The extended safe-life methodology uses a combination of flaw tolerance and damage tolerance approaches. Component replacement times are calculated using the traditional safe-life approach with full-scale fatigue tests on specimens with flaws representative of defects that occur in manufacturing and in service. However, in addition to establishing failure, the tests monitor crack growth to failure. Damage tolerance principles then establish inspection periods for cracks and to evaluate degradations in fatigue or quasi-static strengths. By this combination of approaches, parts can not only be retired when loss of function occurs but component service life can be increased when validated by inspection.

The flaws intentionally inflicted on specimens are the maximum probable flaw sizes determined from historical data from manufacture and service. In order to establish inspection procedures, damaged areas must be accessible during service, and cracks must be detectable and measurable by the method and tools that will be used to perform this inspection. The accurate determination of the initial damage size and its propagating length must take into account the experience and training of the personnel who will be performing the inspection. This

approach, therefore, promotes user-friendly designs for operators and maintenance institutions.

In order for this methodology to provide even higher reliability and to significantly reduce accidents, components and sub-assemblies can be designed with multiple load paths and to provide easy access for inspection in critical locations where tests have indicated that damage will occur. As a standard procedure, the extended safe-life approach establishes replacement or retirement lives of principal structural components. Components are designed to be tolerant to flaws, to propagate cracks slowly, and to provide redundant load paths in critical structures. These features, together with planned inspections to ascertain that crack sizes are below acceptable limits for components to perform their stated functions, will assure that components will successfully react the spectrum of operating loads until the next inspection period. This approach is applicable to metals and composites.

AIRFRAME SUBSTANTIATION PROCEDURES

Metal and composite airframes are designed to meet crashworthiness and ballistic-tolerance requirements. These static requirements establish the static design criteria where the static limit loads exceed the operational oscillatory loads by a large margin. Rotor-generated oscillatory loads in helicopter airframes are also significantly below the loads from these static requirements. Low-cycle airframe fatigue loads from normal landings and maneuvers, although higher than rotor-generated oscillatory loads, are still well below the static design criteria.

Metallic structures, with high stress concentration areas, generally have low fatigue endurance limits compared to their static ultimate strengths. Composites, on the other hand, with flatter fatigue S-N curve shapes and low fatigue sensitivity to stress concentrations, tend to have relatively high fatigue endurance limits in tension-dominated modes compared to their static ultimate strengths. Experience has shown that a composite airframe with good static strength will have high fatigue strength margins. This means that a fatigue test of a full-scale, composite airframe may not be necessary provided that analysis, based on a building block approach, validates that the oscillatory loads do not exceed the endurance limits.

Full-scale fatigue tests of the airframe must be conducted when detail analysis cannot be corroborated by test in highly loaded areas or when the load path is complex. In these cases, the test may be on the full airframe or a major sub-assembly of the airframe. Occurrences of a large number of high oscillatory loads, large out-of-plane loads, highly loaded complex joint configurations in major bulkheads or when the effects of loads are not known are all reasons for conducting full-scale airframe tests. Additionally, any airframe designed for repeated

heavy lift missions are candidates for full-scale airframe fatigue tests.

In the manufacture of the helicopter airframe, a large number of flaws may be permitted in order to reduce the costs of production, inspection, and rework. Since composites are inherently damage tolerant, any damage-tolerant features in airframe design only enhances the fail-safety of composite construction. A "no-growth" qualification on this basis requires a low strain level that further reduces the possibility of generating fatigue damage or propagating flaws.

The presence of a crack or a flaw in an airframe structure does not preclude it from being airworthy. However, in order to demonstrate airworthiness, a full-scale test under representative loads may be necessary. Cracks in metal structures or delaminations in composite structures may be acceptable if the flaw growth rate or the rate at which new cracks appear in adjacent structures are deemed inconsequential to the overall structural integrity.

As an example, the fatigue test on the all-composite tailboom for Boeing's MD 500N helicopter produced two unexpected benefits. Very early in the tests, cracks developed in an area subjected to high out-of-plane bending loads. The locations of the cracks confirmed the high strains predicted by finite element analysis. The growth of the cracks was monitored and found to be arrested after an additional 225 simulated hours of flight. No further growth was measured after 4,100 simulated flight hours and two applications of enhanced limit load. The test validated the crack to be benign and detectable, and was used to establish a safe inspection interval, Ref. 12. A field repair was then designed and tested on the same test article for the duration of the fatigue test. The repair procedure was qualified for field application and inspection procedures established.

A damage tolerance approach to the design of helicopter airframes is almost always chosen, and composite construction makes this an even more advantageous choice. The full-scale fatigue test must take into account the material and operational variability while demonstrating structural integrity. The variability may be demonstrated by (1) a test conducted under environmental conditions; (2) applying a scatter factor to the fatigue test time; (3) multiplying the test loads by a load enhancement factor (LEF); or (4) a combination of all three.

The full-scale fatigue test for Boeing's MD 500N tailboom was conducted under hot-wet conditions using an LEF of 1.18 and a scatter factor of 2 on life. The tailboom was soaked for 30 days at 85 percent relative humidity and a temperature of 71°C before the test. If a lower load enhancement factor were selected, the test duration would have been longer. If the failure mode is unknown or there is more than one failure mode, the highest applicable LEF must be used in order that all possible failure modes are considered. However, the selected LEF must not be so large where high deflections

result in false failure modes. In tests of metal airframes, LEFs are generally not used. When testing a composite or a composite-cum-metal airframe, a high LEF must be carefully chosen in order to avoid qualifying an over-designed structure or recommending early retirement or repairing the airframe during the test. The airframe must be analyzed in detail for overloads and high local residual stresses in order to select the appropriate LEF. This methodology is based on that developed for fixed-wing aircraft, Ref. 13.

The size of the airframe for the full-scale test may make moisture and temperature conditioning impractical. A combined test and analysis procedure is then used. The full-scale static test article is instrumented at critical locations where the measured strain can correlate the finite element analysis. Once the correlation between the test and the finite element model has been established, the maximum strains for all the critical loading conditions can be calculated. The maximum strains are then compared to the material allowable data to validate the structural integrity of the airframe design.

AGING AIRCRAFT ISSUES FOR FIXED-WING AIRCRAFT AND HELICOPTERS

As stated in the introduction, aging aircraft concerns were brought to the public's attention when the Aloha accident occurred in 1988. This type of problem could be expected in the civilian fixed-wing fleet because of the number of aircraft operating and the many hours that these aircraft fly. Although structural aging problems could be thought of as those relating to years in service, they can also be attributed to the cumulative effect of cyclic loading during operation. From the context of structural integrity, the fixed-wing fleet has identified corrosion and wide-spread fatigue damage (or, multi-site damage) as the two primary aging aircraft issues. The civilian fleet has an advantage over the military fleet in that replacement of older aircraft is more likely than in the military fleet where, since the end of the cold war, the military has continued to cut back their acquisitions, and aircraft are now expected to remain in service for as long as 50 years (see Table 1).

Although the FAA addressed the aging problem issues as early as 1968, Ref. 14, with new procedures issued in 1978 for maintaining the safety of aircraft as they age, it was not until the Aloha accident that a national program was developed to study in detail the problems associated with aging. Along with several conferences that focused on aging issues and several prominent research programs that have come into existence, an Aging Aircraft Task Force (AATF) was formed. An independent advisory panel, Technical Oversight Group for Aging Aircraft (TOGAA), was appointed to continuously review the aircraft industry and airlines. TOGAA began their review of the helicopter civilian fleet in 1994. At one of the industry and FAA review meetings in 1995 the helicopter industry stated that "aging is not a significant

issue for rotating parts because they are replaced or extensively refurbished on a periodic basis" as a result of the safe-life design philosophy. Even though TOGAA agreed with this statement, the helicopter community would consequently state that issues such as corrosion and multi-site damage are fatigue phenomena that do occur in helicopters.

Corrosion is a problem that obviously occurs on most metal structures that operate in a salt or moisture environment. As to the second primary aging phenomenon of multi-site damage, known as MSD in the fixed-wing community, different experiences seem to exist depending on the operation of the helicopter. If MSD does occur it is often more an economic concern for helicopters in repairing these multiple damages rather than a safety issue. Quite the reverse is the case with fixed-wing aircraft. While there are some obvious differences in aging problems between helicopter and fixed-wing aircraft structures, there are also some similarities. These differences and similarities on corrosion, MSD, structural inspections, loads monitoring, and the separate problems associated with military and civilian operator are discussed below.

Corrosion

It is obvious that corrosion is a problem for both the fixed- and rotary-wing aircraft since both are still predominately metal structures. Since many aircraft operate around a salt water environment (U.S. Navy, U.S. Coast Guard, and helicopter operators of off-shore oil platforms in the North Sea and the U.S. Gulf Coast), the corrosion problem requires constant vigilance in both preventing corrosion and repairing corrosion damage. Currently no mathematical model exists that can predict the rate of accumulation of corrosion. In fact because of the many variables that effect the accumulation of corrosion (mostly where the rotorcraft is being used), it is probably not possible to predict the accumulation of corrosion without a usage monitoring system such as a Health and Usage Monitoring System (HUMS) unit. With a usage monitoring system capable of tracking corrosion as it accumulates, an on-board computer could predict the life of a fatigue crack propagating in the corrosive environment. The technology to predict fatigue crack growth in a known corrosive environment exists today.

As is the case with the fixed-wing community, a computer database that reflects all of the experiences of rotorcraft in a corrosive environment is not available today. The U.S. Coast Guard has developed a limited database two years ago. In fixed-wing military operations where the U.S. Air Force has been tracking structural problems through their Aircraft Structural Integrity Program, ASIP, since the early 70's, a database on corrosion problems has been recorded prior to 1990 for only the KC-135 aircraft. The KC-135 has been in operation for 40 years, and there is no obvious date for it to end service, which is another example of how military

aircraft will continue to age. Since the military had not planned on keeping aircraft in service for such extensive periods, aging considerations were not considered in their original designs. Also, many of these aircraft were designed in the 1950's with materials that are more susceptible to corrosion than currently available materials. This problem is illustrated in the case of a main rotor grip on an "older" helicopter that was made of 2014-T6 aluminum alloy, Fig. 7. After corrosion problems occurred, the material was changed to 7075-T73 aluminum alloy which offers greater resistance to stress corrosion, Ref. 15. One design concept that occurs in all airframe structures that lead to corrosion is the lap splice joint in fuselage skin construction. The area of the lap joint has been shown to initiate and accumulate corrosion. The lap splice joint problem can be selectively, though not completely, eliminated in helicopter airframes. The joint is more extensively used in fixed-wing aircraft.

One weakness that can cause problems in helicopters that does not occur in most fixed-wing aircraft concerning corrosion is the safe-life design methodology. Traditional safe-life does not account for any deviation in fatigue strength that may occur over time (aging) due

to flaws or corrosion that develop during manufacturing, maintenance or service. However, in 1988 for helicopters the flaw tolerance design methodology was added to the Federal Air Regulation to help alleviate this short-coming of the traditional safe-life design method. Several approaches can be taken to modify the safe-life methodology after the helicopter has entered service. These are illustrated below.

In the case of the structural life management of a horizontal hinge pin for the CH-53 A/D helicopter, which had been an out-of-production U.S. Navy helicopter, a study of the conditions that can reduce structural integrity was undertaken in order to extend the life of this component, Ref. 16. The horizontal hinge pin is made of 4340 steel and had experienced corrosion problems during its service life. Using flaw tolerance concepts a coupon test program was developed using a "worst case" corrosion pit. The coupon specimens were tested in fatigue and showed a 63% reduction in fatigue strength, Fig. 8. This study set the inspection interval at 1,200 hours, which coincided with a scheduled overhaul and which allowed regular inspections up to its normal retirement time of 8,300 hours.

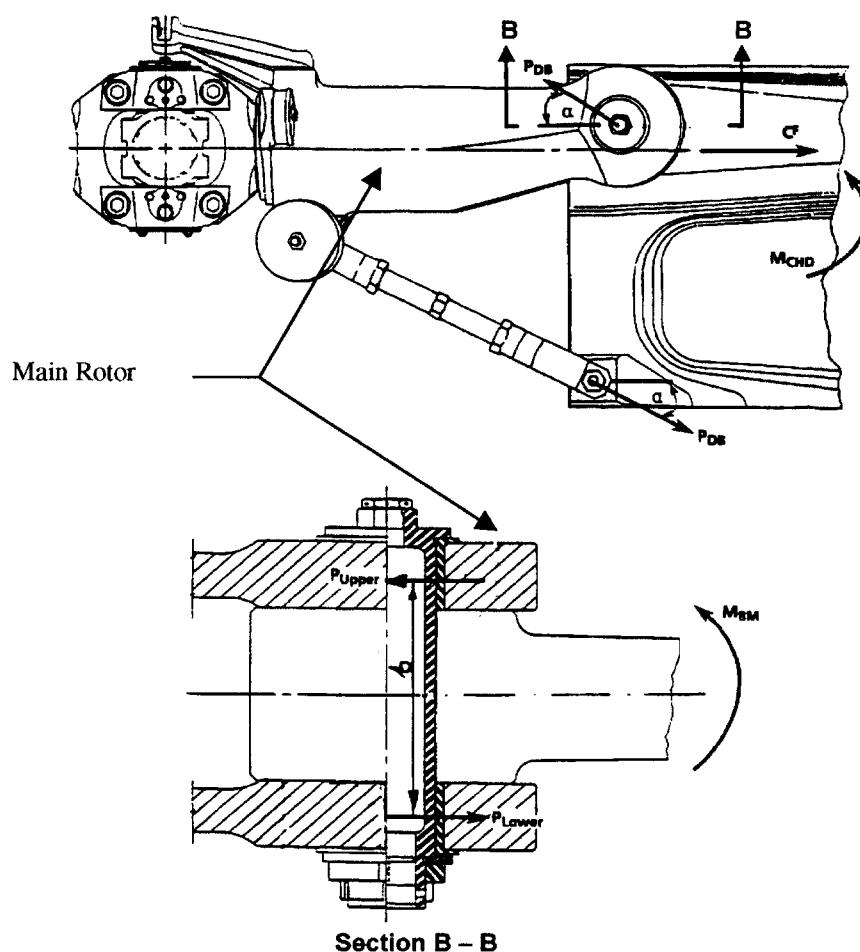


Fig. 7 Main rotor grip where corrosion was mitigated by replacing the aluminum alloy (Ref. 15)

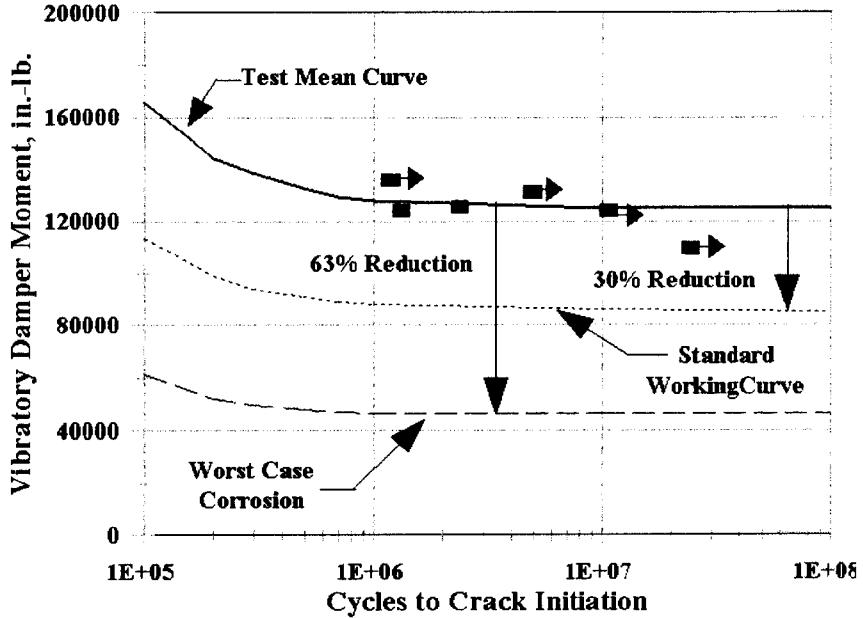


Fig. 8 Effect of corrosion on the S-N curve of the CH-53 A/D horizontal hinge pin (Ref. 16)

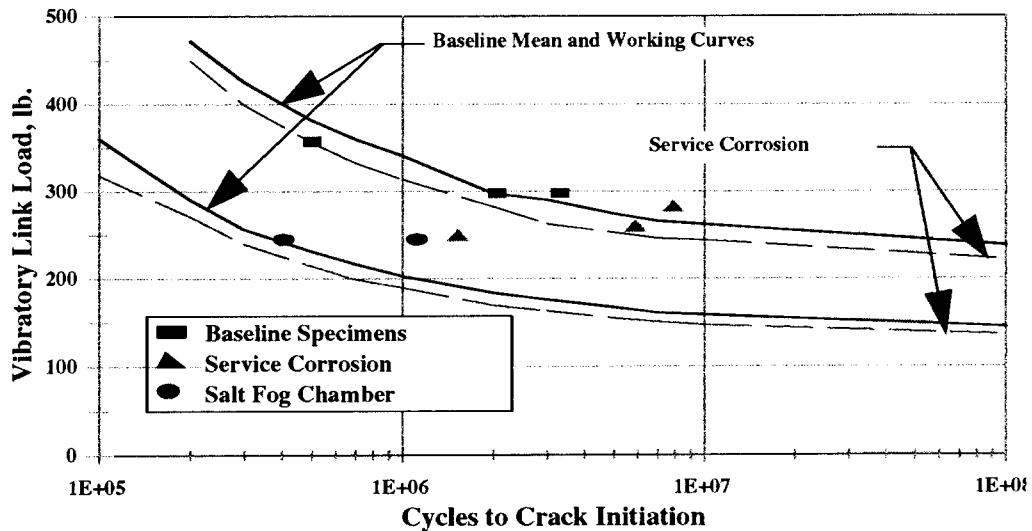


Fig. 9 Effect of corrosion on the S-N curve of the S-76 tail rotor pitch horn (Ref. 16)

A second example was the tail rotor pitch horn of the S-76 helicopter, Ref. 16. This component also experienced corrosion. The pitch horn is made of an aluminum alloy with a 22,000-hour retirement life. A "worst case" condition was used to evaluate failure due to corrosion, Fig. 9. In this case, instead of using inspections to monitor the structural integrity of the pitch horn, its retirement life was reduced to 12,000 hours. This decision was based on the fact that some corrosion had been seen to occur on all pitch horns that had been inspected.

Where corrosion is a problem, extreme vigilance must be exercised to prevent the problem from becoming severe and requiring extensive repairs. Evidence of this severity has been seen in the civilian fixed-wing fleet. This situation is often aggravated when aircraft, subjected to short-term changes of ownership or operators, do not appear to receive adequate maintenance.

Multi-Site Damage

Multi-site damage, also called wide-spread fatigue damage, occurs in structures with "similar details operating at similar stresses where structural capability could be affected by the interaction of similar cracks."

In fixed-wing aircraft typical structures where MSD could occur would be wings and empennages with chordwise splices and rib-to-stiffener attachments. This type of damage is also common in fuselages with lap splice joints. Generally in the fuselages of fixed-wing aircraft and helicopters, with all the rivet holes, lugs, and other fittings, MSD is very likely to occur in several locations. MSD is less a safety issue in helicopters than for fixed-wing aircraft because of the pressurized cabins found in the transport civilian fleet. When a series of small cracks link up to form one large crack in the structure of a pressurized cabin, the extra energy released could be a serious safety hazard that would not occur in the non-pressurized fuselages of helicopters. This was the situation in the structural damage suffered by the Aloha aircraft in 1988 where the safety of the aircraft was jeopardized.

It appears that even though MSD has not been a safety issue in helicopters, general cracking is observed more and more. This is one area of concern where regular inspections can be of great value. In Reference 7, Eurocopter estimates that about 20 accidents and major incidents over 43 million flight hours could have been avoided by periodic inspections of the helicopters. If fleet surveys of problems are systematically recorded, trends can be identified and major structural failures can be avoided in the future. If multiple cracking is found early through inspection, modifications can be made and catastrophic conditions can be preempted from developing. The military believes that helicopter airframes are not "tracked" as well as the dynamic components. Since airframes are the structures where MSD is most likely to develop, some type of tracking (inspection) program should be instituted to preclude much larger and expensive solutions later in service.

One example where a cracking problem like MSD has been noted is in the Royal Navy's EH-101 helicopters, Ref. 17. During an extensive fatigue test program cracks were noticed to form in the rear fuselage where conventional skin-stringer designs are used. This part of the airframe was identified as one to be managed on an "on-condition" basis. This is a good example where a thorough initial fatigue test program has identified MSD and developed a good inspection program to extend the safe service life of the EH-101 helicopter.

Programs are now being instituted to monitor areas where future problems may occur and take action early to mitigate them and extend component service lives. A case in point is the U.S. Navy's Helicopter Integrated Diagnostic System (HIDS). One aspect of HIDS is to track vibration in helicopters in order to maintain smooth rotor blade operations thus reducing fatigue-type loads and alleviate cracking problems like MSD.

Loads and Usage Monitoring

Loads and usage monitoring is one of the key technical topics of the day. It makes little difference how exact the life prediction model is if the magnitudes of the loads are

not known or are inexact. When two different pilots fly the same maneuver on the same helicopter the measured loads can differ by as much as a factor of two. When a syllabus of typical maneuvers were flown by six pilots on the same helicopter, the average coefficient of variation in measured loads for all maneuvers was between 7 to 10 percentages. Consequently, in order to include "unknown" loads in the loads analysis for calculating the safe retirement life, only the top-of-scatter loads are considered. This assumes that the maximum load measured in a maneuver occurs throughout the maneuver. However, the loads experienced by a structure not only varies from aircraft to aircraft but also on how an aircraft is flown. Thus, part retirement based on average aircraft usage is difficult to quantify. Again, as aircraft age, the operation of individual aircraft are increasingly and substantially different from the average of the whole fleet, and estimates of the retirement lives of components based on average aircraft usage are inadequate. The deterministic loads and allowable used in safe-life methodology has also been attributed for the frequent inspections recommended by manufacturers. A probabilistic approach has been suggested for a more efficiently managed fleet.

As it was previously noted under the section on MSD, a system like the U.S. Navy's HIDS can play a role in loads and usage monitoring. Certainly with the rapid advancements in algorithms, sensors and computer technology smaller and more sophisticated usage and loads monitoring systems are now available, and it is expected that an on-board life prediction system will soon be available to record actual loads for the usage monitoring system. This will account for the variation in loads caused by the pilot's input. In the case of the HIDS system, the use of an automated diagnostic system for helicopters has been shown to provide early warning of damage and wear before the occurrence of failure. This has been demonstrated by the U.S. Navy on the drive train components of the UH-60 helicopter using HIDS, Ref. 18. The HIDS system is designed to detect early, and monitor the progression of, incipient "fault condition." Thereby, the health of a component can be known at any point in time, the accumulation of damage can be tracked and component replacement at normal overhaul times on aging aircraft can be effectively managed. Thus, HIDS not only maintains the helicopter's structural integrity but also improves availability and reliability while minimizing cost through scheduled maintenance.

The variation in usage from aircraft to aircraft occurs not only with changing requirements of the operating agency but also when the aircraft is operated by several agencies. This is the case of the European multi-nation Tornado aircraft, a total of about 1000 of which have been procured by three nations through 1997. It was originally conceived as a low-level strike and reconnaissance aircraft. With three different nations using this aircraft, the Tornado is now a multi-role

aircraft and loads are accumulated at rates different from the original design spectrum. This example is typical of the dilemma faced by all aircraft manufacturers.

Nondestructive Inspection and Reliability

Perhaps the most important aspect of assuring structural integrity in aging aircraft and yet probably the weakest link is that of identifying and locating damage through the use of nondestructive inspection (NDI) methods. If the service life of an aircraft is extended, damage is inevitable, which must be detected reliably and repaired properly. Often, it is not a matter of how small a damage can be located, but how reliably can such small areas of damage be found. For most metal structures, this damage is either corrosion or a crack. The question in NDI is not how small a crack can be detected, but how large of a crack can be missed from being detected.

In the technology of managing structural life through inspections the U.S. Air Force has the most experience since its formal adoption of the ASIP program when MIL-HDBK-1530, Ref. 9, was published in 1972. In regard to a crack size that can be detected with a high degree of reliability, the Air Force in its damage tolerance design philosophy uses a rogue flaw (the largest flaw likely to exist) of 1.27 mm as its standard in fixed-wing aircraft. In regard to how small a crack must be found in helicopters to insure their structural integrity, crack sizes of the order of about 0.4 mm are often quoted. While some sources suggest that eddy current can locate cracks of these sizes fairly reliably, others have found that the smallest crack that can be detected with a high degree of reliability is 0.8 mm. Reliable crack detection is of prime importance as the rotorcraft community attempts to move towards a life management system based on extended safe-life of its structures. In the current environment of mostly safe-life designs, the rotorcraft community is limited in its ability to manage aging helicopter structures because the Palmgren-Miner's rule in safe-life methodology does not physically model the initiation and growth of a crack. Some sources even believe that a 0.4 mm initial crack size is too large for damage-tolerant designs of helicopter components and that crack sizes as small as 0.2 mm must be the design basis. What is important is reliable detection of any crack in the prevailing environment under which the inspection is conducted. As the helicopter community continues to design affordable structures to damage tolerance and to stringent weight requirements, it is obvious that increasingly small cracks are required to be reliably detected. To increase the reliability of detecting cracks, automated inspection systems are required to eliminate the human error in reading and interpreting results. Nondestructive inspection methods and acceptance/failure criteria are described in greater detail below.

The Aging Problem of Military Aircraft

From statements made above and as shown in Table 1, the military aging problem is much more severe than that of the civilian fleet. In the case of military aircraft, it is becoming increasingly difficult to justify the budget to regularly replace the military fleet. The military has also not been able to-date to develop a database of aging aircraft problems in order to identify the time and rationale when structural parts of a helicopter should be replaced.

In the U.S. Army, which has a fleet of about 5,000 helicopters, the Aircraft Condition Evaluation (ACE) program was recently initiated to re-engineer older helicopters to as-new condition. This program has already revealed that while new helicopters have time between overhauls of 1,000 hours or more, older, refurbished parts often have only a few hundred hours between overhaul. It is, therefore, not cheaper to maintain older helicopters than to buy new ones because evidence shows that the cost of maintaining helicopters rises continuously with the age of the helicopter. A case in point is the U.S. Army's CH-47D Chinook helicopters which are re-engineered and refurbished from the CH-47A. The increasing cost for maintaining the CH-47D helicopters over a period of nine years is shown in Fig. 10, Ref. 1. As the hours of operation of a helicopter increases, the military budget is doubly penalized with higher operation and support (O&S) cost and higher maintenance cost because the helicopter is further aged. The O&S cost for CH-47D helicopters in terms of flight hours is shown in Fig. 11, Ref. 1.

The U.S. Army continues to maintain its helicopter fleet with a safe-life design philosophy. If cracks occur, the problems are often managed using a damage-tolerant inspection procedure. However, if the component that is experiencing these problems is redesigned, the service life is again calculated on the basis of the safe-life approach. The Army is considering adopting more and more damage tolerance types of procedures, but the cost and reliability of NDI remain the principle barrier for wider use of the damage tolerance methodology. The remoteness of some deployment sites and the use of the foot soldier for inspections are two reasons why the Army considers the damage tolerance approach to be too risky at present.

The U.S. Navy also has an aging fleet of helicopters, with the average age of about 19 years, Fig. 1. The Navy expects to alleviate the aging issue through an aggressive replacement program starting in 2005; however, the rate of acquisition will depend on the available budget. The Navy's procedures for managing their fleet are almost the same as the U.S. Army's structural life management philosophy. That is, the safe-life methodology sets retirement times. In case of failure, damage tolerance principles are used to address problems in the field while a safe-life redesign is undertaken to increase the service life of the component, Ref. 3. The U.S. Navy is not too encouraged with the damage tolerance approach because

of the difficult environment in which it operates, and because shipboard inspection is often undertaken by seamen without extensive experience, Ref. 2.

The U.S. Air Force, while desiring to move towards a damage tolerance philosophy for helicopters because of the successful application of its ASIP program to its fixed-wing fleet, has depended on the Army for its structural integrity methods because its fleet of helicopters is small.

NDI AND ACCEPTANCE/FAILURE CRITERIA

Overview

It has been discussed above why application of damage tolerance principles are being inhibited by the cost and reliability of NDI methods. This section describes the NDI methods for composite structures; the same principles are generally applicable to metallic components. Composite structures are more difficult to

inspect because they are made of non-homogeneous materials and are manufactured by a variety of processes each with its specific requirements for quality. In addition, special attention must be given to composite structures in order that internal defects or damage can be detected and assessed. In metallic materials, flaws are modeled and linear elastic fracture mechanics applied to predict failure. Unfortunately a unified failure model is still under development for fiber-reinforced composite materials. Experimental methods for detecting internal flaws in composite structures and comparing the flaws to reference standards are the only means for evaluating the structural integrity and the residual level of performance in safety-critical applications.

Non-destructive inspection methods identify manufacturing and in-service defects in structures without degrading their quality or affecting their serviceability. Defects can be external and internal to the structure. External defects can be visually inspected, such as dimensions, finish, and warpage. Internal defects

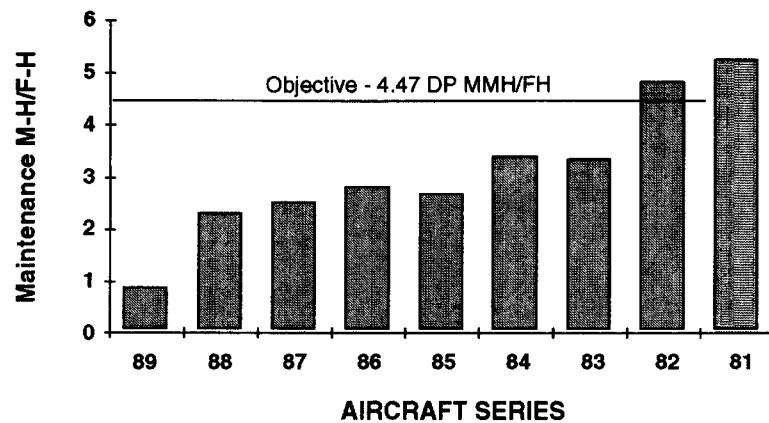


Fig. 10 Maintenance man-hours per flight hour for the Chinook helicopters in the U.S. Army's inventory (Ref. 1)

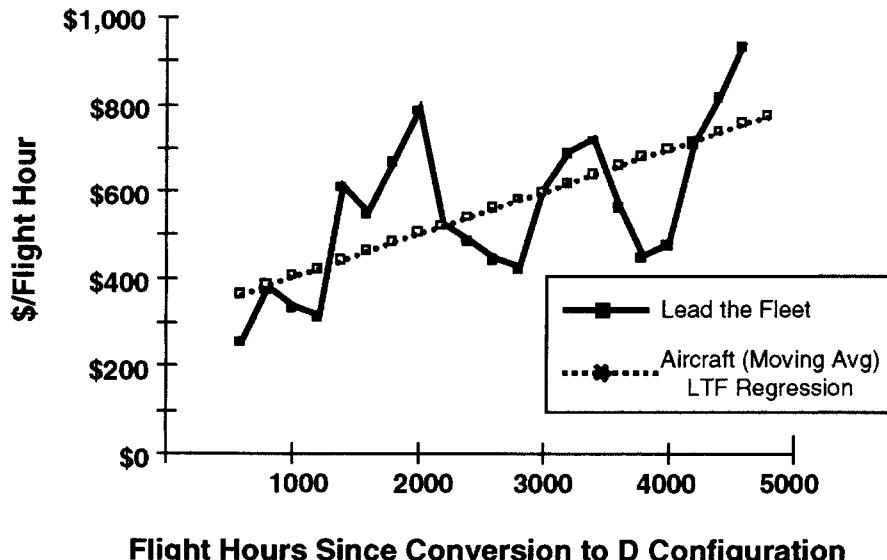


Fig. 11 The O&S cost of Chinook helicopters with respect to hours of operation (Ref. 1)

of most concern in composites are delaminations, inclusions, voids, resin-rich and resin-starved areas, fiber misalignments and breakage, and debonds. NDI is part of quality assurance that controls the manufacturing process in order to meet design specifications, produce repeatable products and reduce cost.

Composite materials are made of two constituents: fibers for strength and a matrix to bind the fibers to shape. Defects will inevitably occur in composite structures. NDI is used to evaluate the criticality of the defect(s), i.e., number, size(s) and location(s). The implementation of NDI must, therefore, begin during design by arranging structural details to facilitate inspection, and with analyses to qualify the arrangements and identify the critical sizes and locations of defects in terms of the capabilities of the available techniques. In order to conduct NDI efficiently and accurately, reference standards must be established which take into account the design, equipment, the minimum defect size that can be detected, types of defects and acceptance criteria. The acceptance criteria are developed through destructive tests and analyses to quantify the criticality of defects in a flawed structure.

Types of Internal Defects

Internal defects typically occur due to variations in the material from batch to batch and due to the variables in the manufacturing process. Most defects can be easily prevented through proper process control, of which NDI is an integral part. The internal defects of most concern for manufacturers of composite structures, and where NDI alone can assess the "damage" caused by these defects in order to assess the residual level of performance are described in Table 3 (Ref. 12).

NDI Methods

The integrity of composite structures is evaluated by identifying (sizes and locations of) defects using NDI methods. Several NDI methods are in use, but none of them quantifies the integrity of the structure. The results of NDI are compared with drawing specifications and reference standards in order to identify the sizes and locations of defects. A structural analysis and test program then quantify the integrity of the structure. Descriptions of NDI methods are available in several publications, including Ref. 19-25. The advantages and disadvantages of the more widely used methods are shown in Table 4 (Ref. 12), and their efficacy in detecting defects in Table 5 (Ref. 12). The most common methods for NDI are visual, radiographic and ultrasonic inspections. Specialized inspection methods are used for specific applications. For example, Cat-Scan has been used to inspect alignment and compaction of fibers.

Acceptance Criteria for Production Parts

Since a composite structure is a complex assembly of elements made from several material types and forms, it

is inevitable that defects of various shapes and sizes will be present in production parts. During the design process the criticality of defects are analytically determined, which then establishes the acceptance criteria. The acceptance criteria vary widely with the type of defect, the structure being inspected and the NDI method used. For example,

- 1) the acceptance criteria are more stringent for primary structures than for secondary structures,
- 2) the acceptance criteria for voids and porosity can be given in terms of signal attenuation for ultrasonic inspection or in terms of linear dimension and area when radiography is used, and
- 3) the limiting area of acceptability can be for individual voids or for an envelope around several scattered defects.

Non-destructive inspection is the joint responsibility of engineering, manufacturing and process control to assure that composite structures are manufactured to a quality consistent with the design requirements. To assure early detection of defects, specifications of acceptance criteria for each NDI method are documented on the basis of the minimum defect size that can be detected. However, these documents are general in their requirements, and specific criteria for primary components or at specific locations are included in engineering drawings. Specific locations may be critical because they are in highly stressed areas or because they are difficult to inspect.

The documents on general acceptance criteria include the acceptance and repairable limits and the repair procedure. If a defect can be repaired, the component is directed to process specifications and then re-evaluated through NDI. Typical acceptance criteria for laminated structures are given in Table 6 (Ref. 12). Since acceptance criteria are based on manufacturing experience and are peculiar to the NDI equipment in use, the list of defects given in the tables will vary and their limits may be refined to the unique capabilities of each manufacturer.

The acceptance criteria for composite sandwich structures include both typical defects as well as defects that are unique to the designs of a manufacturer. The design details are especially important for composite components of the rotor system, e.g., the blades. Since the blade is constructed to meet the specifications of static and fatigue strengths, stiffness, and the distribution of weight, specific details of blade elements - the spar, leading edge, trailing edge and after-body - may be separately identified in the acceptance criteria. The number of details identified depends on the manufacturing method used for the blade elements and the assembly procedure adopted. Typical acceptance criteria for defects in sandwich structures are given in Table 7, Ref. 12.

Failure Criteria for Development Test

Composites under inspection will always indicate structural defects either visually or by NDI. These defects have to be identified as structurally significant before further substantiation is undertaken. As stated previously, composites are inherently flawed, with some level of each of these flaws acceptable at any given location of the component. By "acceptable," it is meant that the structural integrity of the component is not compromised by the presence of the flaw.

The declaration of a structural failure then must be postponed until it is verified that the observed defect can grow to be unstable to the point where structural integrity

is affected. In safe-life testing, it may be appropriate to consider the initiation point as the number of test cycles where the defect was first observed or detected. If the defect growth is arrested and component structural integrity is not reduced, the defect can be declared as "cosmetic" and ignored in analysis and in service.

In extended safe-life testing, the initial detectability must be defined in visual terms or in terms of one of the specific methods in Table 4, which then becomes the standard method for inspection in service. The rate of growth of the defect is then monitored in the test, and the equivalent flight hours to unstable crack length is used to calculate the in-service inspection intervals.

Table 3. Descriptions of Defects in Composite Structures (Ref. 12)

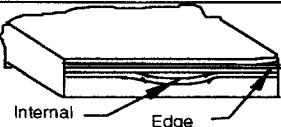
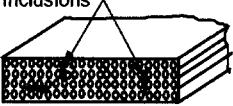
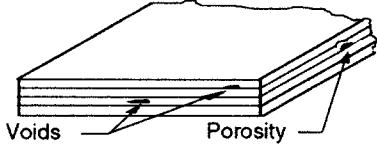
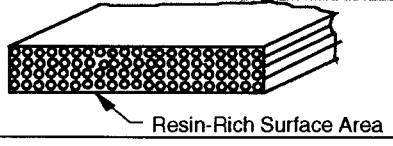
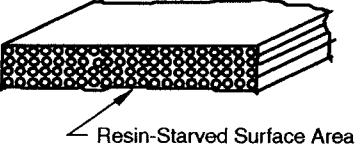
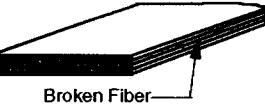
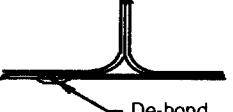
Defect	View	Description
Delamination		Delaminations are separations within plies of a laminate, and caused by improper surface preparation, contamination and embedded foreign matter.
Inclusions		Inclusions are foreign matter embedded in and between laminae.
Voids and Porosity		Voids and porosity are entrapped air and gas bubbles, and are caused by volatile substances, improper flow of resin and unequal pressure distribution. Voids are clustered in the resin, while porosity are pockets within the solid material..
Resin-Rich Area		Resin-rich areas are localized, and filled with resin or lacking in fiber. This defect is caused by improper compaction or bleeding.
Resin-Starved Area		Resin-starved areas are localized with insufficient resin evident as dry spots, or having low gloss or where fibers are exposed. This defect is caused by improper compaction or bleeding.
Fiber Misalignment, Wrinkling, Buckling		Fiber misalignment is a distortion of the plies resulting in changes from the desired orientation, or in fiber wrinkling and buckling. These defects are due to improper lay-up and cure.
Fiber Breakage		Broken fibers are discontinuous or misplaced fibers due to improper handling or lay-up.
De-bond		De-bonds occur between different details of the built-up structure. Lack of bonding is due to contamination of the surface, excessive pressure or bad fit.

Table 4. Efficiencies of NDI Methods for Composites (Ref. 12)

NDI Method	Application	Advantages	Disadvantages
Visual	1. Surface defects	1. Simple 2. Economical	1. Limited information 2. Surface defects, finish
Coin Tap	1. Large voids, debonds and delaminations	1. Simple 2. Economical	1. Limited information 2. Difficult with very large structures.
Radiographic	1. Internal defects in sandwich structures 2. Edge delamination and damage	1. Easy to use 2. Equipment readily available 3. Extensive information on the state of damage 4. Permanent record	1. Radiation safety concerns. 2. Relatively expensive 3. Penetrant required 4. All damages (thin flaws perpendicular to the beam) may not be detected.
Ultrasonic (C-Scan)	1. Voids and porosity 2. Delaminations	1. Very accurate results 2. Relatively low-cost process 3. Permanent record 4. Thick structures 5. Can be automated	1. Slow operation 2. Couplant medium needed 3. Bulky, relatively expensive equipment 4. Difficulty with complex geometry (core, etc.)
Acoustic Ultrasonic	1. Delaminations and voids 2. Fiber breakage 3. Fiber-matrix interface 4. Fiber orientation	1. Higher resolution than C-scan. 2. Quickly evaluates for acceptance or rejection 3. Permanent record.	1. Limited to small areas 2. Dependent on surface geometry since the medium is the structure
Thermographic	1. Delaminations and voids 2. Debonds	1. Simple system 2. Quantitative results 3. Real-time images 4. Surface contact not required.	1. Experience required to size and type of defect 2. Limited size of structure 3. Thin specimens only
Acoustic Emission	1. All defects	1. Continuous monitoring 2. Global	1. Structure is under load 2. Voluminous data 3. Simple geometry
Holographic	1. Delaminations and voids 2. Core-to-skin de-bonds	1. Rapid evaluation 2. Relatively inexpensive 3. Surface preparation not required 4. Permanent record 5. Real-time image	1. Sensitive to vibration if not coupled 2. Optically accurate alignment required 3. Laser safety concerns
Shearographic	1. Delaminations and voids 2. Impact damage 3. Cracks in holes 4. De-bonds	1. Rapid evaluation 2. Relatively inexpensive 3. Portable equipment 4. On-aircraft inspection 5. Permanent record 6. Real-time image	1. Laser safety concerns
Eddy Current	1. Surface defects	1. Relatively low-cost 2. Can be automated	1. Limited to good electrical conductors
Edge Replication	1. Surface defects 2. Initiation and progression of cracks	1. Ply-by-ply record 2. Saturation number of transverse cracks 3. Simple, rapid 4. Permanent record	1. Surface defects only 2. Often limited to coupon-type specimens

Table 5 Capabilities of NDI Methods for Composites (Ref. 12)

NDI Method	Core Damage	Delamination	De-bond	Fiber Break	Fiber Misalignment	Impact Damage	Inclusion	Resin Variation	Void
Visual		S	S	S	S	S		S	S
Radiography	A	C	C	C	B	B	A	A	A
Ultrasonic	B	A	A	B	B	A	A	A	A
Acoustic Ultrasonic	B	A	A	A	B	A	A	A	A
Thermo-graphic	C	B	B			B	C	C	B
Acoustic Emission		A	A	A		A			
Holographic	A	A	A	B		A	B		A
Shearographic	A	A	A	B		A	C		A
Eddy Current ^a	B	B	B	A		B	B	B	A
<u>Legend -</u>		A: Good detection C: Detection when defect is large				B: May not detect minor damage S: Detection at surface or edge			
Notes: ^a : For good electrical conductors									

Table 6 Typical Acceptance Criteria For Laminated Composite Structures (Ref. 12)

No.	Discrepancy	Acceptance Limit ^a (in.)	Repairable Limit ^a (in.)	Repair Procedure ^b
1	Surface Depressions	0.25 dia., 0.03 deep or less than 25% of laminate thickness.	>0.25 dia, <0.05 deep and no fiber damage.	Sand, clean, fill with epoxy, cure, sand to dimensions and verify.
2	Surface Pin Holes	0.10 dia. or 0.03 dia. holes over 10% of laminate area.	<0.25 dia. or >0.03 dia. covering 70% of laminate area.	Sand, clean, fill with epoxy, cure, sand to dimensions and verify.
3	Surface Cracks	None.	Not applicable.	Not applicable.
4	Resin-Rich	0.03 thick.	None.	Sand (avoid fiber damage), clean, verify dimensions.
5	Resin-Starved	All, if only on surface ply.	None.	Sand (avoid fiber damage), clean, brush epoxy, cure, sand to dimensions, verify.
6	Frays, Burrs	0.13 at machined edge.	>0.13 or affecting assembly.	Trim, apply adhesive, cure, sand to dimension, verify.
7	Surface Inclusions	0.10 sq.in. each with 5% of laminate area.	Greater than acceptable limit.	Sand, clean, fill with epoxy, cure, sand to dimension, verify.
8	Warps	0.01 gap from flat surface with 10 LB applied every 12 inches.	None.	Not Applicable.
9	Delaminations, Radii Bridging	0.125 sq.in., and away from other indications ^c .	(Dimension varies with distance from edge.)	Drill 0.0625 or smaller holes, inject epoxy, clean, cure, sand to dimension, verify.
10	Porosity, Voids	3%-10% depends on class of structure ^d .	None.	Not applicable.

Notes:

^a The limits shown vary with manufacturers and can be overridden by specifications on engineering drawings.^b The repairs are conducted to process specifications. Only the outlines are presented here.^c Several limits might be included to reflect importance of structure (primary, secondary, redundant, etc.) or NDI method used. Limits can also be given in terms of attenuation of the signal.^d The limits vary with the importance of the structure and the requirements of the certifying agency. Specific limits are often stated on engineering drawings.

Table 7 Typical Acceptance Criteria for Honeycomb Sandwich Composite Structures (Ref. 12)

No.	Discrepancy	Acceptance Limit ^a (in.)	Repairable Limit ^a	Repair Procedure ^b
1	Core Distortion	Must be correctly in place in lay-up tool when tacked.	Unlimited for Nomex cores None for fiberglass cores.	Re-form Nomex cores and assemble. Not applicable for fiberglass cores.
2	Core Crushing	(Depends on location. Usually) 1 cell deep, 1.00 in any direction.	None.	Remove and replace.
3	Core Buckling	1.00 in any direction and 1% buckled of total depth, with next buckled area at least 6.00 away.	None.	Remove and replace.
4	Core Nesting	(Number of cell rows depend on particular location.)	None.	Remove and replace.
5	Bond Line Thickness	1 layer: 0.003 to 0.015 2 layers: max. 0.020, etc.	None.	Not applicable.
6	De-bonds	Must be continuous.	Unlimited only where repairable; otherwise none.	Repair where possible; otherwise cut out, replace, re-bond.
7	Foam Adhesive	Generally a gap length of 3-4 cells over 6.00; but depends on particular location.	Unlimited only where repairable; otherwise none.	Repair where possible; otherwise cut out, replace, re-bond.
8	Discontinuous Bond Line Transverse to Splice	0.1 visible width.	None.	Not applicable.
9	Foreign Material in Bond Line	None.	None.	Not Applicable.

Notes:

- ^aThe limits shown vary with manufacturers and can be overridden by specifications on engineering drawings.
- ^bThe repairs are conducted to process specifications. Only the outlines are presented here.

SUMMARY AND CONCLUSIONS

The helicopter industry has been aware for sometime of the importance of addressing the issues of aging aircraft. The methods for substantiating the structural integrity of components leads to possible approaches for addressing problems of aging aircraft. These methodologies for assuring the structural integrity of structures have been discussed in detail. With the multiplicity of materials, manufacturing processes and operating spectra, the extended safe-life approach appears best to address issues of both initial qualification and aging aircraft. The extended safe-life approach though appropriate for civilian operations is not fully favored by the military. The military prefers the safe-life approach for logistical reasons and because of difficulties in using NDI methods under operational conditions.

A discussion of the available NDI methods has been provided in terms of acceptance criteria to help identify problems associated with NDI and to provide directions for alleviating the difficulties encountered by the military. The severe conditions under which the U.S. Navy has to operate is shown in Fig.12, and some times under these conditions, damage tolerance evaluations are conducted by seamen, the average age of whom in the

current U.S. Navy is 19 years, Ref. 2. Similarly, the evaluations by the foot soldier in the U.S. Army may be conducted under less than optimum, though a different set of difficult, conditions.

The large number of accidents caused by operational and environmental conditions beg further attention because these exceed accidents due to structural failures by several magnitudes. To reduce these accidents, further



Fig. 12 Operating conditions of the U.S. Navy (Ref. 3)

development of all-weather helicopters are required, so that helicopters are equipped with such features as fly-by-wire and health and usage monitoring systems.

The technology for analysis exists today to assure structural integrity issues of aging helicopters. The extended safe-life approach encompasses the best of several methodologies to make the qualification of structural integrity affordable. In conjunction with qualification assurance, the development and validation of simple-to-use health, structural and usage monitoring systems will further improve the integrity of structures to meet the increasingly stringent requirements of both the civilian and military operators, and will also reduce accidents due to non-structural causes.

Another aspect that has been highlighted is the need for a database which should be a systematic record of the failures experienced and of inspection results of the causes of degradation and failures. This database will provide validation that manufacturers require for further improving the structural integrity and for providing optimum guidance to helicopter operators. With innovative designs, advanced manufacturing processes, improved NDI techniques and a systematic database on experiences encountered, quality assurance for helicopters can eliminate structural failures in worldwide, all-missions operations.

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Aging Aircraft Subsystems

Equipment Life Extension within the Tornado Program

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Abstract

Restrictions on nations military budgets require that Air Forces will operate high sophisticated weapon systems such as military aircraft far beyond their original designed life. Modernization, life extension and aging aircraft programs are defined to ensure that future mission requirements can be fulfilled and the airworthiness of the aircraft is maintained until the out of service date.

Whilst the operational life is increasing, various aging effects take place and impair the structural and functional integrity of high aged aircraft subsystems even before their design life are reached. An integrated process was defined between customer and industry with the aim of extending the operational life of the aircraft under limited budgetary conditions and tight timescales.

Under the scope of this task, aging of equipment and components, the impact of aging problems on structural and functional integrity, the impact of customer extended operational requirements on equipment limitations and the consequences of age related failures on flight safety have been assessed.

Recommendations for revised maintenance and inspection programs and revised limitations have been or will be worked out by industry for safety and non-safety relevant equipment to enable formal certification of the aircraft for the extended period.

This work is generally accompanied by equipment design review, review of the in-service usage, investigation of reported in-service problems and related statistics, master inspections of subsystems in high aged aircraft and investigation of aged loan equipment, provided by the Air Forces. Additionally, programs are conducted to establish revised in-service operational conditions.

Various programs have been defined by the FAA and US Air Force, which aim on addressing safety issues in aging aircraft structural and non-structural systems.

For the 20 years old Tornado aircraft, a life extension program for structure, subsystems and engine is presently being carried out, whose results should enable certification of the aircraft for the extended

operational phase and support safe operation of the aircraft for the next 25 years which means a life extension by roughly one aircraft life.

This paper discusses the various aspects related to aging of subsystems and the content and philosophy of the Tornado equipment life extension program. Furthermore specific problems identified in various subsystems are presented together with a description of the way ahead for life extension.

Overview

This lecture comprises of four major parts:

- Part 1:** Introduction and a general review of the aging system problematic and other problem areas related to an aged aircraft
- Part 2:** A description of the subsystem life extension program for the Tornado IDS variant
- Part 3:** Subsystem Life extension approach and experiences
- Part 4:** Conclusion

Part 1

Review of the aging system problematic and other areas of concern

Introduction

When aircraft are getting older operators are increasingly confronted with problems, which are caused by the aging process to which the aircraft are exposed. Aging could degrade the integrity of structures, equipment and other components to such an extend that the consequences of failures could be catastrophic.

In the civil aviation very spectacular accidents occurred where aging was or probably was the dominating factor through which the accident was caused:

- the Aloha Airlines 737 accident where due to hidden corrosion in the lap splices the aircraft lost a big portion of the upper front fuselage during flight
- the TWA Flight 800 in July 1996, where the explosion of the center fuel tank might be caused by wiring problems.

Both accidents revealed that

- existing maintenance procedures are not adequately addressing aging problems in areas, which are inaccessible for visual inspections
- the effects of aging on structural and functional integrity of aircraft subsystem components and flight safety could be catastrophic

To operate flight systems safely within their certified life, maintenance procedures have been defined taking into account their fail-safe design philosophy. This means that any failure within the subsystem, which could lead to a safety critical hazard, would be extremely improbable and fly to failure is the prevailing philosophy. This is a major reason why the On-condition Maintenance Concept (OCM) is applied to the majority of subsystem components.

As aircraft age, the number of failures due to aging in functional equipment or other system components increases. Aircraft availability and mission success will be increasingly impaired due to unscheduled maintenance actions required.

Under the view of declining budgets and the necessity to operate aircraft beyond their design life investigations are performed in order to review the aging process of subsystem components in aging aircraft.

But it is important to recognize, that the aging process of any component cannot be prevented but the velocity of aging can significantly be reduced by e.g.

- identifying critical areas and components where aging could have serious consequences
- reviewing the current aging process of subsystem components in high aged aircraft or of high aged components
- reviewing the current maintenance procedures and policy regarding the aging problematic
- improving preventive maintenance actions in order to address aging in an early stage.

Fig. 1 presents the current age of the Tornado IDS fleet. It indicates that the oldest Tornado aircraft have already reached a service life of 20 years.

Considering a further planned usage beyond the year 2020 for the IDS variant, some IDS aircraft will stay in service for nearly 50 years.

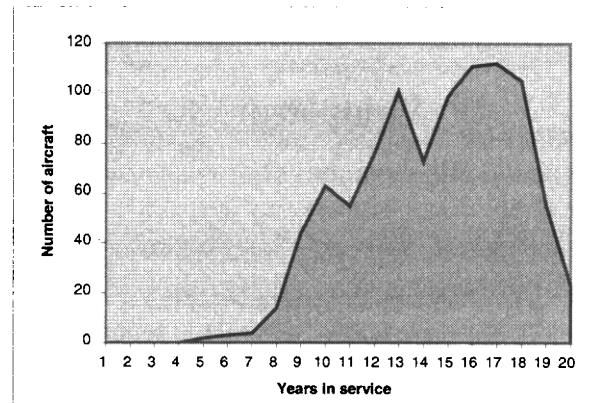


Fig. 1: Age of the Tornado IDS fleet

This, however, does not mean that the equipments and components of subsystems will remain in service such a long time: it is very probable that many equipment and components of systems have usually already been replaced during regular or unscheduled maintenance by overhauled, repaired or new items. A uniform in-service life for all subsystem components in an aged aircraft does not exist.

It is therefore important for critical equipment to record in-service operational and maintenance data in order to be able to assess the usage history and life consumption and define the point of retirement or where maintenance actions or overhaul is required to ensure safe operation..

Certification rules require replacement of equipment when its certified life is reached. As system components have often a life potential which is bigger than the certified life and in order to get the maximum benefit of equipment in terms of performance and costs, an subsystem life extension program is carried out.

Within this life extension process, which will be described in a later chapter, aging problems within each subsystem have been investigated and taken into account for defining appropriate measures for life extension.

The following chapter discusses briefly which effects are generally contributing to aging of subsystem components.

Aging mechanisms

Aging is understood as a process, where the structural and/or functional integrity of equipment / components will be continuously degraded by the exposure to environmental conditions, under which the equipment is operated.

This could lead in the worst case to a situation where the aged equipment / component cannot fulfill

any more its designed function, even before the design life of the item is reached. In this case, subsystem functions, which the item has fulfilled or supported, could be lost or degraded, which may also affect flight safety.

There are various mechanisms, which alone or in combination are responsible for equipment aging.

- Exposure to normal or salty atmosphere, heat, water, oil, grease, fuel, ultraviolet light, etc., which could lead to corrosion, embrittlement, swelling, overheating, melting or other material degradation, electrical interruptions, short circuits, etc.
- Exposure to vibration and acoustic environment, which could lead to fatigue damages, wear and tear, etc.
- Endurance, which could lead to leakage, wear and tear, etc.
- Storage, which means exposure to storage conditions, specified by the equipment supplier or in-service practice
- Maintenance activities, which can induce accidental damages (which could be a particular problem for wiring)

It is likely that reliability and availability of equipment will be impaired by these mechanisms.

Additionally, incorrect installation of equipment could also have a detrimental effect and support the aging process, e.g. enable chafing of wire bundles on surrounding structure (this can be the result of poor design but also of poor maintenance).

Aged equipment and components will be removed from service if further use is not recommended for safety reasons or if further safe operation is uneconomical due to required inspection and maintenance activities.

Life expired equipment can be defined by the fact that any in (equipment) specification specified and certified life limitations are reached by the in-service usage and therefore need to be replaced.

Life extension might be possible and needs to be investigated. In case, the design and certification authority permit further on aircraft operation under clearly defined conditions, further use of the equipment can be tolerated without formal re-qualification and for a limited time: as the aircraft runs out of certification (>4000 FH) and the formal certification process up to 8000 FH has not been completed yet, the definition of such conditions and requirements for further usage up to 5000 FH is part of the Tornado life extension program.

Influence of maintenance on aging

As already mentioned, aging of equipment and components cannot be prevented but slowed down. It is reasonable to say that if maintenance is poorly or not timely conducted certain aging effects (e.g. wear and tear, contamination, corrosion, etc.) could be accelerated or even started.

This means that the physical condition of aged equipment and components in different aircraft can vary due to different quality of maintenance carried out which could result in earlier retirement of components than assigned.

Aircraft maintenance programs will always be a compromise as beside others logistic costs and overall fleet management are important issues, which drive their definition.

The Tornado is operated by three different air forces, which have different maintenance programs and procedures. The major depot inspection for the RAF and GAF is currently at 2400 FH (or 14 years, RAF only) and 2000 FH for the IAF. Considering the fact that at the beginning of the Tornado program the DI was at 1600 FH, the time where a thorough investigation of the aircraft condition is performed has significantly increased (by 50%).

In view of this and to address aging problems, it is important to include preventive maintenance actions for critical equipment and areas, at shorter intervals in periodic servicing schedules or in special inspections, to ensure that aging problems will be detected in an early state where the effort for repair is still acceptable and an airworthiness critical situation far away.

Within the Tornado subsystem life extension program aging effects on equipment and components are investigated as far as they are obvious and known. The results of these investigations will be used by the manufacturers to establish revised maintenance procedures for their equipment where necessary.

Other problems areas

Besides the aging problematic, other topics which are not related to aging but causes problems in an aged aircraft need to be considered and investigated. The list might not be complete but describe the major problem areas.

- Operational conditions

These might have changed during the lifetime of the aircraft.

For an aged aircraft where life extension is an issue the current and future planned operational conditions need to be investigated and compared to the original design requirements. This can be done

by engineering assessments or if the conditions cannot be established on an theoretical basis by Operational Load Measurement (OLM) programs.

The outcome of such programs will be used to calculate existing clearances and life potential or provide a basis for further qualification testing.

The establishment and conduction of OLM programs are often very costly and time consuming. Therefore, any decision for such programs should be taken early enough to ensure that results are available in time.

In the Tornado program such Operational Load Measurements (OLM) are conducted in the Landing Gear System as presented later.

Usage history, life consumption

As required by formal certification rules, equipment, which exceeds stated life limits are formally out of certification. Therefore, the usage history needs to be established and this requires the availability of complete in-service documentation (LOG cards, maintenance documentation).

Unfortunately, it has been experienced that it is sometimes very difficult to get all the information necessary to assess service life and life consumption. For safety relevant and life limited components this causes many problems as it might be not possible in these cases to establish whether an equipment is already life expired or not.

Repairs and concessions

The influence of repairs and concessions on life limitations needs to be assessed.

Normally, minor concessions should not affect life limits, but if the operational conditions have changed in the past, the classification of concessions needs to be reassessed.

Obsolescence

Obsolescence is a problem, which becomes more problematic with increasing age of the aircraft.

Materials and components, which have been used in the original design, could be no more available and this could require redesign and requalification activities if no proofed and certified alternatives exist.

Especially for electronic parts obsolescence is a critical and costly issue as technology progress over the last 20 years was so enormous that certain items remain only available in a limited quantity. A strategy for future support of electronic equipment needs to be established.

Availability of original supplier

For most of the major equipment there was only one supplier selected, who has designed and qualified the item. This supplier might have disappeared or no more able to produce the required component. Introduction of alternative supplier will cause problems in terms of time and costs. Dependency on one supplier is critical but usual.

Costs for procurement of equipment

The procurement of equipment for replacing aged or life expired equipment after 20 years in service could be very costly as e.g. original supplier are no more existent, design and qualification of new components are necessary or the required number of equipment is very small.

Inspections and inspection efficiency

With the aging of the aircraft aging effects will increasingly require inspections of critical equipment and components to ensure that any aging effect will not lead into a safety hazardous situation. The success of inspections is very dependent on the selection of the appropriate methods, availability of required equipment, clearly defined inspection procedures, availability of trained personal.

Maintenance documentation

Existing maintenance procedures, which do not address aging as experienced by investigations or sampling, needs to be revised. Requirements or recommendations of the equipment manufacturer, defined during the life extension process, should be included.

Failure and maintenance reporting systems

Most aircraft users are maintaining comprehensive databases to collect and evaluate failure and maintenance reports for the whole aircraft down to equipment and component level in order to identify unreliable equipment and other problem areas.

However, in service data collection systems are mainly defined by the aircraft users and reflect their specific needs and points of interest. The definition of data elements and the level of detail may significantly vary, even between the different users of the same aircraft type.

For this reason, the probability, that failure and maintenance data collected during service are compatible with design analysis and predictions is fairly low.

Moreover, in service data are not always available to the system design authority at a level required e.g. for equipment aging investigations.

The value of an in service database can only be as good as the entries. It is therefore important to consider that any analysis or curves derived from databases need specific interpretation and should be taken with care.

The major "lessons learnt" for future data collection systems are therefore continuous availability to both user and manufacturer and the need for compatibility with design analyses, in order to gain maximum benefit over the aircraft lifetime.

Similar problems are mentioned in other publications. It is therefore important to consider that any analysis or curves derived from databases should be taken with care.

Aging of Tornado subsystems

To get an overview on the magnitude of subsystem aging problems, the increase of aging related failure rates have been investigated, Fig. 2.

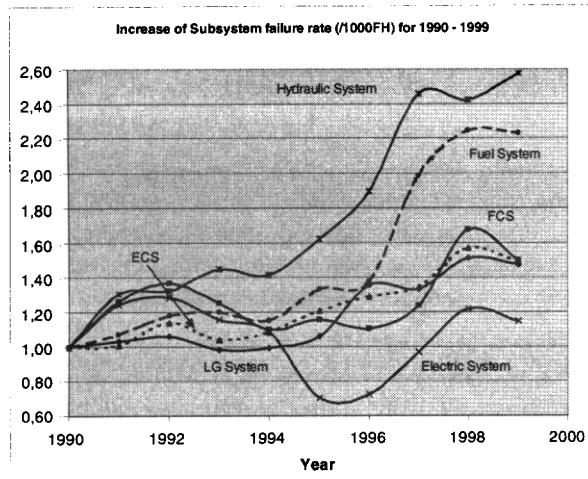


Fig. 2: Reported in-service failures due to aging

For these trend curves failures are considered which could be attributed to aging, such as corrosion, wear and tear, leakage, contamination, porosity, brittleness, contact damage, isolation damage, etc.

However, it should be noted that these curves are only trend curves as it is very difficult to analyze whether the reported failures are really due to aging or other influences.

The analysis has been performed for the following subsystems and for the period 1990 until 1999:

- Landing Gear subsystem

- Flight Control subsystem
- Electric Subsystem
- ECS Subsystem
- Fuel Subsystem
- Hydraulic Subsystem

The investigation show an significant increase of aging related failures over the last decade for the

- Hydraulic Subsystem by 160%
- Fuel Subsystem by 120%
- Landing Gear, Flight Control and ECS subsystem by 50%
- Electric Subsystem by 15%.

The biggest contribution to the above curves in the Hydraulic and Fuel System is leakage with 43%, respectively 19%.

After the description of general aspects of the aging problematic and additional problem areas, we have encountered in the Tornado program, part 2 will describe the Tornado Subsystem Life Extension program for the Tornado IDS variant, as it is presently being carried out.

Part 2

Tornado subsystem life extension program

As introduction a very brief survey of the Tornado program should be given, before describing in detail the philosophy and process of the Tornado Subsystem Life Extension program.

Tornado Program



The Tornado aircraft was designed and developed in three European countries, the United Kingdom (42,5% share), Germany (42,5%) and Italy (15%) according a set of tri-national agreed Performance and Design Requirements (PDR) which form a compromise between national requirements.

The major aerospace companies in these three countries were in charge to build this aircraft, which have been British Aerospace, BAe, (now BAE SYSTEMS), Messerschmitt-Bölkow-Blohm, MBB, (now DaimlerChrysler Aerospace, DASA) and Aeritalia (now Alenia).

The PANAVIA Aircraft GmbH was formed by these three companies (PANAVIA Partner Companies, PPCs) in order to manage the program on industry side.

On the customer side, the program is controlled by NAMMO, the NATO MRCA Development and Production Management Organization, and today managed by the NATO European EF2000 and Tornado Agency, NETMA (former NAMMA). NAMMO and NETMA are staffed by specialists from all three Nations.

The first flight of Tornado was successfully performed in 1974 and the first production aircraft was completed in 1980.

In total 977 aircraft have been built in two major variants, that are the

- Interdiction Strike/Attack (IDS) variant, which is operated by all three Nations Air Forces (RAF: 228 aircraft, GAF: 357, IAF: 99) and also the Royal Saudi Air Force (RSAF) (96)

and the

- Air Defense Version (ADV), which is operated by the RAF (173) and the RSAF (24). Today also the IAF operate ADV aircraft, leased from the RAF.

Single stick and trainer versions exist for both variants. Additional, for the IDS various sub-variants, e.g. the Electronic Combat Reconnaissance (ECR) variant, exist, differently equipped to perform specific missions and to carry specific external stores.

The development and production of Tornado was shared between the three Nations roughly according the intended number of aircraft they wished to procure.

The responsibilities for design and production of the airframe were shared between BAE SYSTEMS (front and rear fuselage), DASA (center fuselage) and Alenia (wing).

For the development and qualification of subsystems, System Design Responsibilities (SDR) for each system and Equipment Design Responsibilities (EDR) were defined. SDRs and EDRs were shared between BAE SYSTEMS and DASA but they were not necessarily in one hand.

As there are some differences in subsystem design between IDS and ADV, a separate life extension task has been defined for the ADV variant. This task deals with the life extension of ADV specific equipment and equipment, which is common to IDS but differentially loaded in service and therefore different life limitations needs to be established.

Common items to IDS with same operational loading conditions are covered by the IDS life extension process.

In this lecture, only the IDS life extension process will be described where the fundamentals of this process have been discussed and defined.

Tornado IDS Subsystem Life Extension Process

The subsystem life extension program is integral part of the aircraft life extension program, which includes also engines and structures. Aircraft certification will be a compilation of the output of all three separate tasks.

Nations requirements for the extended use of Tornado IDS are shown in Fig. 3.

The basic tri-national requirement for life extension was that PANAVIA should establish the basis for aircraft certification up to 8000 FH, based on the original PDR requirements.

Additional to that, UK wish to extend the life of their IDS for a further 2000 FH up to 10000 FH.

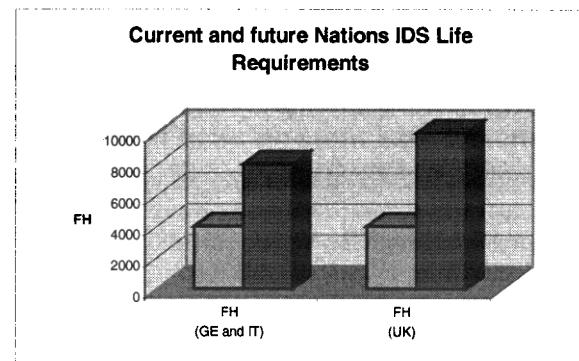


Fig. 3: Life extension requirements

The current planning of Nations air forces revealed that they intend to operate the IDS Tornado aircraft beyond the year 2025 (GE and IT) and 2020 (UK), which means that the oldest aircraft will stay in service for about 50 years.

Certification documentation of the aircraft and its equipment includes life limitations. Consequently, operational use by the air forces beyond this limitation is at Air Forces own risk and absolves PANAVIA and their supplier from any product liability.

Therefore, Nations via NETMA required that PANAVIA should be tasked to perform work that would allow PANAVIA to resume the product liability for the extended period.

In 1995 PANAVIA was contracted to carry out a program, which aims on extending the life of TORNADO IDS and ECR variants significantly beyond the current life limits up to 8000 FH, which means a doubling of the aircraft operational life.

The overriding constraint upon extended certification is that no significant reduction of flight safety is acceptable, although equipment reliability in some cases may be degraded with time. This requires the identification and assessment of equipment and components which failure would be safety critical.

The certification of equipment for the extended period requires that the equipment *Declaration of Design & Performance* (DDP) will have to be updated stating the new life limits as outcome from the life extension process.

The approach of identifying equipment, which should be regarded for life extension activities, considers that aircraft subsystems are generally designed to fail safe design principles and thus comprise of equipment and components which can fail to perform their function without presenting or significantly contributing to a hazardous situation.

The philosophy of this approach was agreed with NETMA and PANAVIA was tasked to focus on safety relevant items which definition for the purpose of this task has been defined as follows:

Components are considered to be *safety relevant*, where a single failure would cause a hazard or which in combination with other failures would significantly contribute to a *hazardous situation*.

A *hazardous situation* occurs when conditions arise which threaten the safety of the crew, aircraft, ground crew or third parties.

For this life extension task the term *safety relevant* covers both "Flight Safety Critical" and "Flight Safety Involved" incidents.

The following list states the Tornado flight systems, which are subject of life extension activities:

- Crew System
- Electric System
- Environment Control System
- Fire Detection & Suppression System
- Flight Instruments
- Flight Guidance & Control System
- Flight Control System
- Fuel System
- Hydraulic System
- Landing Gear System
- Pitot Static System

- Propulsion System
- Secondary Power System
- Weapon delivery system
- UK Armament Control System

The assessment of the safety relevance of equipment attachment structures was also part of the task. Each EDR performed the assessment of attachment structures for his area of responsibility and defined in conjunction with the relevant structure SDR appropriate measures for life extension, if necessary.

For financing and managing purposes, the whole life extension task was structured into 4 different phases, in which the level of involvement of the customer, PANAVIA and the PPCs and the supplier varies significantly.

Phase A: Identification of Safety Relevant Equipment and Attachment structures

Phase B: Nations review of Phase A recommendations and selection of items which should be subject of further life extension work

Phase C: Equipment extended certification activities

Phase D: Aircraft extended certification and preparation of a new PANAVIA Lifed Items List.

It is currently planned to finish the IDS subsystem life extension program at the end of 2002 including UK IDS life extension to 10000 FH and aircraft certification activities.

Phase A

Identification of Safety Relevant Equipment and Attachment structures

A process flow diagram for Phase A and B is given in the annex.

For each system, review of the existing *System Safety Assessments Reports* was the agreed approach to determine the safety or non-safety relevance of equipment according to their contribution to a hazardous situation.

As these reports have been issued a considerably time ago during the design phase of Tornado, new experiences and in-service arisings have also been taken into account as well as the effects of equipment failure on surrounding area and also where failure effects cross the boundaries of different subsystems.

Completeness was ensured by use of equipment lists (WUC) and illustrated part catalogues.

As a result of the safety assessments during Phase A, a *Flight System Hazard Analysis Report* has been produced by PANAVIA covering separately each subsystem.

For each safety relevant item a proforma was written by the Equipment Design Responsible (EDR), which provides informations of the equipment as for instance part number, DDP, qualification status, life limitations and overhaul requirements, its function within the subsystem, failure mode and hazard contribution. The proforma format ensured consistent information for all equipment and components for all subsystems. The overall responsibility of the content of the proforma rests with the SDR for each subsystem.

This report also provide a subsystem equipment list supporting the aircraft level certification for all items which are no more further considered as they are of negligible or low risk.

The final phase A report was subject of a system review meeting attended by NETMA, Nations and PANAVIA in order to further refine the safety relevance of equipment into one of the subsets of safety relevance as described below:

CAT1: Item Safety Relevant, qualification life limitations are critical and to be observed

CAT2: Item Safety Relevant but qualification life limitations are unlikely to be critical and further qualification or analysis to extend life should be considered.

CAT3: Item Safety Relevant, failure or malfunction of the item has no significant airworthiness consequences.

The general guidelines for determination of the safety relevance are presented in the following table.

Severity of hazard	Catastrophic	Critical	Marginal (Major)	Negligible
Single failure Safety critical	CAT1 OR CAT2	CAT1 OR CAT2		
Double failure common mode contribution	CAT1 OR CAT2	CAT1 OR CAT2	Not Safety Relevant	Not Safety Relevant
Double Failure Dormant Item	CAT1 OR CAT2	CAT1 OR CAT2	Not Safety Relevant	Not Safety Relevant
Double Failure Significant contribution / known problem	CAT1 OR CAT2	CAT1 OR CAT2	Not Safety Relevant	Not Safety Relevant
Double Failure Non-Dormant Item	CAT3	CAT3	Not Safety Relevant	Not Safety Relevant
> OR = Triple Failure condition	CAT3	CAT3	Not Safety Relevant	Not Safety Relevant
	Safety relevant		Non Safety Relevant	

This guideline chart cannot be regarded as definitive. It essentially required system specialist engineering judgment to establish the severity of any failure or failure combination consequences.

Additional, the following criteria have been considered in the categorization refinement review:

- Level of resulting hazard
- Probability of occurrence
- Subsystem design philosophy
- Dormancy
- Prevailing poor reliability
- Fatigue, endurance or any other life-limiting factor
- Impact on existing overhaul or maintenance

All components, which have not been categorized in one of the above categories, have been defined as *Non Safety Relevant (NSR)*.

After each subsystem review, each proforma were reviewed and amended to include statements declaring the agreed safety relevant categorization together with the justification argument or evidence.

It was agreed at the end of this phase that equipment, which has been categorized as CAT3 or NSR, could be regarded as suitable for continued service in accordance with an On Condition lifing policy (OCM), which means a concept of maintenance that is free from scheduled actions. Maintenance action is taken only when an item of equipment is found to be defective. Exceptions to OCM are permitted only when a scheduled action is necessary to prevent a safety hazard, inability to meet safe life design criteria, or on grounds of economy to preclude expensive damage.

There are specific items, which need special consideration in terms of life extension:

PANAVIA Standard Items

There is a big number of standard parts as connectors, screws, relays, switches, clamps etc. used all over the aircraft and commonly in different subsystems.

These items are normally not limited to flying hours but are specified to operations, cycles etc. There are also numerous items for which no life limitations exist.

The general approach for standard items is to reduce the list to items for which a life limitation was stated and which is not sufficient in view of the lifing requirements of this task. Additionally, items are considered which contain any degradable or perishable materials.

For these items an assessment on their safety relevance in above terms has been conducted and a

categorization defined. This approach is shown in detail in a flow chart in the annex.

All other standard items, which are not safety relevant, are considered to remain on-condition.

Electrical Wiring

Electrical wiring is largely used in the aircraft. It comprises of PANAVIA wire of various construction and materials, electrical connectors, loom supporting devices etc. The life extension strategy for wiring is discussed in Part 3 of this paper.

Attachment structure items

The system engineers have reviewed the criticality of attachment structure items. It became obvious that not all structure items need specific actions for life extension.

Attachment structure items are generally designed to static stress conditions, loads resulting from crash and maneuver accelerations and stiffness requirements. Additional, the installation uses good design practices related to bend radii, hole edge distances and minimizing tension cleat loading. These criteria result in equipment supports, which are sufficiently robust for any normal aircraft lifting.

It is further considered that fatigue is not a dominating factor for

- Large bore Pipes in the air and Fuel subsystem
- Small bore pipes in the Hydraulic and Oxygen subsystem
- Flying Control Linkages

Whilst good design practices were used in order to reduce the risk of failures, in-service maintenance programs and inspections have to ensure that the integrity of such components due to corrosion, contamination, foreign damages etc. are not impaired.

Attachment structure items, which fall under the above, are no more subject to fatigue life extension activities.

All other components and specific equipment mounting structures are covered by the structure life extension task.

In Phase B now, the results of Phase A activities, laid down in the *System Hazard Analysis Report* formed the basis for the customer review of safety relevant items.

Phase B

Nations review of Phase A recommendations and selection of items which should be subject of further life extension work

During this phase NETMA and Nations reviewed the recommendations provided by PANAVIA within the *System Hazard Analysis Report* with their own specialists and took decisions on which safety relevant component they wish to support for extended certification.

The following criteria (beside others) might have been used by the customer for their decisions:

- Premature failures predominant in-service, i.e. the item does not generally achieve its current design life
- Financial and economic aspects as life extension will be too expensive or too long, low cost item, economic of overhaul, obsolescence arisings
- Modification is already planned to replace existing item

At the end of this phase the customer stated to PANAVIA which equipment should be subject of further life extension activities in Phase C.

Phase C

Equipment extended certification activities

A process flow diagram for Phase C and D can be found in the annex.

This phase consist in detail of

- requesting suppliers proposal for life extension of safety relevant CAT2 equipment which have been selected with a declared preferred option for life extension
- assessment of suppliers proposal and agreement on a technical and financial issues
- issuing the proposal to the customer awaiting authorization for start of work
- conducting of life extension technical work based on original specification requirements
- update of equipment DDP reflecting the latest qualification status and finally
- update of the System Hazard Analysis Report at conclusion of work within a specific subsystem

Phase D

Aircraft extended certification and preparation of a new PANAVIA Lifed Items List.

After completion of the equipment extended qualification activities, this Phase D covers the aircraft level certification activities.

Also within this phase, a new PANAVIA Lifed Items List will be issued, containing

- lifing requirements for all safety relevant items, which have not achieved full qualification evidence (with respect to the LE lifing requirements)
- mandatory overhaul requirements for the extended period

Part 3 will now describe in some detail for various subsystems the agreed specific approach for life extension and major findings and problem areas related to aging experienced during the life extension work.

Part 3

Subsystem Life extension approach and experiences

Life extension measures

Most of the Tornado equipment were only qualified according the life requirements stated in subsystem and equipment specifications and only some equipment have already been subject to extended qualification. For many equipment, where no significant failure at the end of their original qualification was found, the final life is unknown.

The definition of the appropriate measure for equipment life extension was made in cooperation with the equipment supplier taking into account all information available from original qualification, in-service experience and future service requirements.

In principle the following questions must be answered:

- a) Is the presently stated clearance still valid and is there any life potential for extended use beyond 4000 FH based on the existing or changed service requirements or under which conditions is the extended use acceptable?
- b) Are there any limiting factors, as for instance aging problems, low reliability, economical reasons, which would not recommend further usage of the item?

The answers on both items will define future usage requirements.

Item a) will generally be investigated by

- review of the original qualification
- assessment of validity of original specified operational requirements, e.g. supported by OLM programs
- analysis
- conduction of re-qualification programs covering aspects such as fatigue and endurance testing (taking into account revised operational conditions, where agreed with the customer)

Item b) will generally be investigated by

- review of the in-service experience (reported failures, repair and overhaul experience, ...)
- review of MTBF data where available
- sampling of aged equipment from service, for which a full service history can be provided, where the consumed life can be established, which have not been overhauled and where the operational usage is representative for the normal fleet usage
- Master inspections of complete subsystems, which take place on one or several different aircraft, which have a service life near to 4000 FH
- Zonal inspections of critical zones, which have been defined for instance by the results of the Master Inspection or by available experience from service

Although difficult to say generally, the following table tries to give an overview on the measures which are conducted for a certain type of equipment / component. (however this does not mean that all measures are conducted in the same time)

Equipment / Component	Life extension measure
Mechanical equipment	Analysis, fatigue test, overhaul
Mechanical / Hydraulic equipment	Sampling, overhaul, analysis, fatigue, impulse fatigue and endurance testing, definition of inspection and overhaul procedure
Electronic equipment	Sampling, definition of inspection and overhaul procedure
Pipework	Master and Zonal inspections, overhaul, definition of maintenance procedure

Equipment / Component	Life extension measure
Hoses (hydraulic fuel, nitrogen)	Sampling, replacement of high pressure and nitrogen hoses
Standard parts	Review, analysis
Wiring	Sampling, Master and Zonal inspections, definition of test concepts, definition of inspection and maintenance procedure
Attachment structure	Review, fatigue analysis, fatigue test

Following, more detail informations are given for various subsystems:

Landing Gear Subsystem

General

Landing gear systems consist of very different type of components, as for instant load carrying structures, electric components (switches, relays, wiring and connectors, circuit breakers), mechanical / hydraulic components (piping, couplings, hoses, valves, swivels, actuators, manifolds), which are necessary to provide various system functions:

- Take off, landing and ground operation capability
- Retraction & Lowering and emergency lowering
- Steering
- Braking and anti-skid
- Arrested landing capability

Major system failure modes are:

- Collapse of gears due to structural failures
- Inability to lower gears by the normal and emergency system
- Loss of steering and braking
- Inability to provide arrested landing capability

Based on these safety critical failure modes, which have been identified in the *System Safety Assessments Reports*, a list of safety relevant equipment and components have been established.

It should be noted that the load carrying structure is designed according *safe life* principles, i.e. a crack free structure during operation is mandatory as no structural redundancies are provided. Therefore these components have been categorized either as CAT1 or CAT2, for which life extension will be assessed.

Other equipments for which life extension is carried out, are for instance amplifiers, valves, servo valves, main wheel brake, hydraulic swivels, foot motors, actuators, locks, nitrogen bottle, anti-skid generator, etc.

Life extension philosophy

The basic philosophy of life extension of landing gear system components is

- Review of original qualification test results
- Theoretical assessments of the fatigue life using existing or in-service OLM measurements
- Extended fatigue tests of structural components
- Extended endurance and impulse fatigue testing of mechanical / hydraulic components
- Investigation of high aged items (sampling) regarding their physical condition after 4000 FH in service and investigation of problem areas
- Establishing of overhaul and/or revised equipment maintenance procedures which take into account aging problems as experienced during sampling and define preventive actions for further safe use
- Re-provisioning, replacement of items whose life cannot be extended

All subsystem components which are critical for the retraction & lowering function as actuators, locks, but also attachment lugs have already been subject to life extension activities and evidence for an extended use up to 16000 FH has been already demonstrated.

Landing gear structure

Life extension of structural components of the landing gear and its back-up structure are of special concern, as it is known that the operational loading conditions have changed during the lifetime of the aircraft, for instance due to the increased aircraft masses, which are operated today.

It was therefore decided some years ago to carry out an extensive in-service operational load measurement program (OLM) to investigate the operational loading conditions on the landing gear back-up structure and the legs with the aim to review existing clearances and establish the basis for future life extension activities.

Two sets of gears were strain gauged and installed in a GAF and RAF aircraft where also the attachment and back-up structure was strain gauged. In total more than 325 flights have been recorded and assessment of the measured data is in progress.

As these aircraft were operated in different squadrons, by different pilots, with different masses and different store conditions, the loading information provides a good cross-section of the operational service loading of the landing gear and its back-up structure in different air forces.

Initial results of these investigations showed that the loads are partially much more severe than originally anticipated.

The influence of these in-service loads on landing structure clearances is a major part of the life extension activities. As the Tornado Nose Landing Gear (NLG) was only tested up to the original life requirement of 4000 FH, it was decided to carry out a new fatigue test taking into account the in-service loading conditions, which take place on MAFT (Major Airframe Fatigue Test specimen).

For the Main Landing Gear structure, a theoretical assessment of the fatigue clearance has started and later this year, when this assessment will have been finalized, a decision on the way forward will be taken, whether or not an additional fatigue test will be required in order to provide the best possible clearance statement.

Landing Gear Equipment

Landing gear equipment life extension base on the topics already described above. Equipment suppliers have defined their requirements by reviewing own experiences of the in-service behavior of their equipment.

Mostly they have required different numbers of high aged equipment from service for investigation of wear and tear, corrosion, fatigue cracks, etc.

Review of the original qualification and test results and the results of the sampling investigations were primarily used to establish whether life extension of the equipment is possible or not. If life extension seemed possible, the method, which the supplier has agreed with the SDR, is applied.

Aging problems on equipments, which have been identified so far (wear and tear, leakage), will be covered by required overhaul at the end of the original design life of 4000 FH before further use up to 8000 FH will be allowed. The required overhaul should also identify the serviceability of that equipment as being the prerequisite for further use.

Aging and age related problems

It has been already shown in the chart (Fig. 2) that the failure rate for aging related failures has increased over the last 10 years by approximately 50%.

Wear and tear and leaking are the major reported failures. For extended use it is necessary that the problem areas are identified and addressed by modifications, inspections and maintenance.

Beside the life extension program, a very thorough inspection for corrosion, wear and tear of

high aged Main and Nose Landing Gears, which were in service for more than 15 years, has been carried out

Corrosion and wear has been found in some areas, for which repairs are defined. But it will be also important to address these areas by inspections and a revised maintenance policy.

Two areas were found which are of special concern as they demonstrate what can happen if maintenance procedures are inadequate in addressing aging effects in time.

These areas are not directly inspectable and would require partially disassembly. As existing maintenance procedures do not require disassembly of the gears to inspect these areas it could happen that the gears are for years on the aircraft or in stock and the condition of components in certain areas, which are not visible inspectable, are not known:

1. Severe corrosion has been found on the outside of the turnable MLG shockabsorber, Fig. 4, which is made of high strength steel 300M, in an area, which is covered by the Main Fitting and belongs to a closed volume where moisture could stay for a long period of time. Outside of this area no corrosion was found on the shockabsorber.

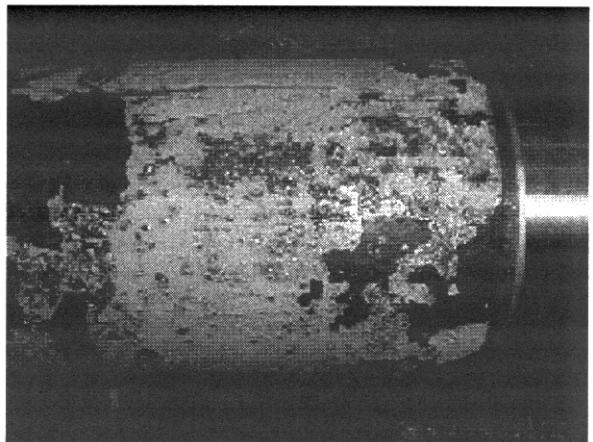


Fig. 4: Corrosion of MLG Shockabsorber

It is considered that corrosion is to be due to localized attacks of the base metal as a result of prolonged exposure to moisture. It is assumed that moisture in this volume is a result of condensation when descending from altitude. The effect of corrosion with regard to stress corrosion and fatigue has been reviewed and not considered critical providing an inspection program is introduced and the component repaired in time. This is important as due to the brittleness of the material 300M, any crack in this area could cause immediate fracture.

An inspection procedure has been issued which address this area in a timely manner.

2. Severe corrosion has also been found at the NLG Steering Collar, Fig. 5.

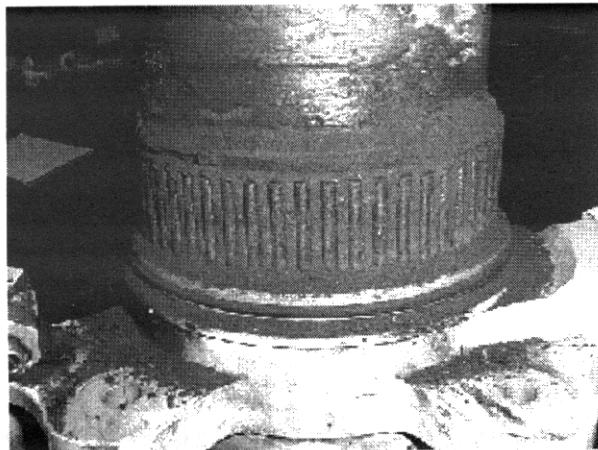


Fig. 5: Corrosion found at the NLG Steering Collar

Access to this area requires also disassembly of the gear. As water was found in this area, it is considered that the sealing becomes insufficient with time. The corrosion became so big that the torque moment required to turn the slider increases far beyond the specified values.

Fortunately, the investigation revealed, that repair is possible, if the repair is applied in time.

Electronic components

All electric and electronic components are sufficient for life extension with the exception of the Anti Skid Control Box, which cannot be life extended due to aging problems due to increasing number of failures of the PCB (printed circuit boards) soldering joints. A repair is not possible because the components and the board is dipped in plastic coating. The exchange of PCBs is not cost effective because the box shelf represents only marginal value of the total box price.

Additional there are other aspects related to the in-service history and built standard of Landing gear components.

Unfortunately it has been experienced that the in-service documentation for landing gear components are not always available. This is considered to be a very critical situation as for instance landing gear structural components are of safe life design with no backup available. The ignorance of service history and consumed life creates many problems, which need to be addressed urgently otherwise the airworthiness is difficult to control.

In case no history can be established any more, a risk assessment has to be performed for each gear and replacement of items before reaching its final life could

be necessary. This situation will become more critical if the life extension activities results in lower clearances for structural components than original stated due to different in-service loading conditions.

Fuel Subsystem

General

The major function of the fuel system is to provide engine fuel supply under all normal operational and emergency conditions.

Two major safety critical system failures as stated in the *System Safety Assessments Report* are

- Insufficient engine fuel supply
- Fire hazard due to fuel leakage

Based on these system failures, equipment and components have been identified whose failure would lead or contribute to these system failures.

Additional, the arrangement of the fuel system within the aircraft fuselage in conjunction with the forced ventilation of compartments / ducts during aircraft operations has been considered for the assessment of the fuel system component safety categorization. Consideration of the general arrangement and operational aspects has revealed that fuel leakage or fuel mist can be carried from the leaking component, which may be located in an area without fire / explosion hazard, into areas with potential ignition sources.

Life extension philosophy

The basic philosophy of life extension of fuel system components is

- Definition of inspection and overhaul procedures for safe pipework operation beyond 4000 FH up to 8000 FH. This will be supported by
 - Master inspections
 - Zonal inspections
- Sampling and re-qualification testing for fuel hoses
- Review of original qualification test results and compare to future operational usage requirements
- Analysis
- Extended qualification by endurance, vibration and pressure cycling tests
- Establishing of future maintenance policy for equipments

Following, (preliminary) results from the life extension work will be presented.

Fuel system Pipework

Fuel pipework is of special concern as fuel leakage due to a broken or leaking pipe or leaking coupling could significantly contribute to safety relevant hazards. The customer requirement for fuel system pipework was that PANAVIA should define inspection / overhaul procedures, which ensure that the safe operation of the pipework is ensured up to 8000 FH.

The procedures will be defined by analyzing the findings gained during *Master* and *Zonal Inspections*, which have been performed on two high aged aircraft from the RAF and GAF (fleet leader) during aircraft depot inspections (DI).

The following activities have been part of the *Master Inspection*:

- visual inspection of all fuel system pipework items, associated fuel system couplings and support / attachment parts
- examination of critical elements, which have been defined prior to the Master Inspection. Special emphasis was given to surface damages, corrosion, connection and attachment areas, dimensional checks
- proof pressure test of each critical element
- X-Ray examination of all welded and brazed joints
- Fluorescent penetrate tests of all welds
- NDT investigation of critical items which experience more than 200 °C during service in order to detect materials alteration due to the exposure to high temperatures
- Examination of seals
- definition of inspection areas and maintenance measures which should be considered for the final pipework inspection / overhaul procedure to be applied on all aircraft beyond 4000 FH

Zonal Inspections, which took place on two RAF and two GAF fleet leader aircraft, based on the findings of the *Master Inspection*. They should confirm the findings of the Master Inspection and provide additional practical evidence for definition of the final pipework Inspection / Overhaul procedure.

Master Inspections have revealed that there were no cracks and no problems with welds. The only problem related to aging was corrosion.

The results of these investigations can be summarized:

- Corrosion was detected on 35% of all pipework items; 14% of these items were classified as "reusable" and 21% as "to be scrapped"

- 40% of all pipework items inspected in the ventral duct area have shown corrosion
- 28% of all pipework items inspected in the spine area have shown corrosion

Fig. 6 shows the propagation of intercrystalline corrosion in the fuel pipe structure (Aluminum) in the marked area.

The findings during the master inspection require inspections and an optimized maintenance policy. Fortunately, the degree of corrosion, which was found on pipes and couplings, need no immediate replacements due to an imminent safety hazard but those pipes which have been classified as "to be scrapped" have been replaced.

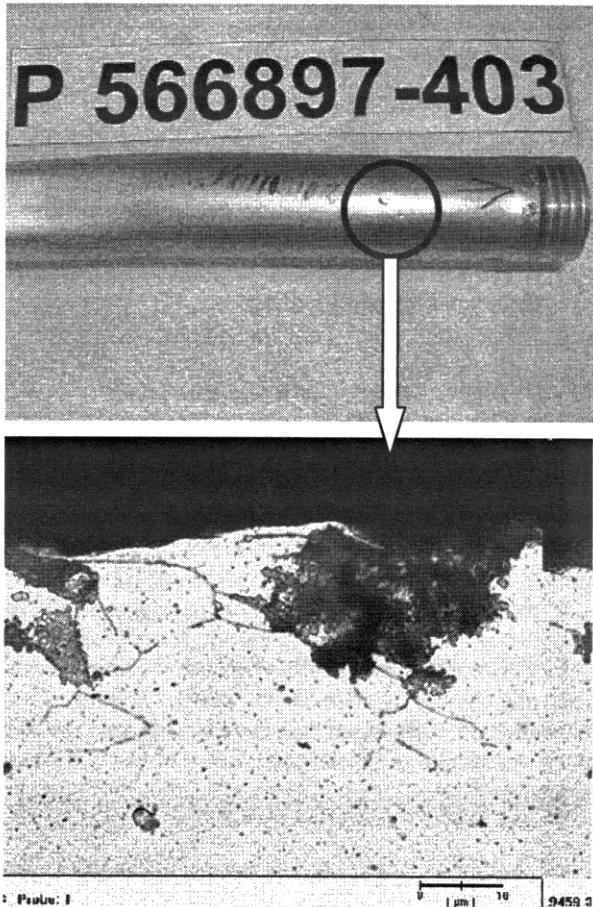


Fig. 6: Corrosion on a fuel pipe and Micrograph Examination of corrosion mark

Fuel flexible Hoses

Flexible hoses using Teflon or rubber have been defined as CAT 1 and they will be replaced after their certified life is reached (Teflon hoses after 4000 FH, rubber hoses after 6 years). This is a preventive measure to prevent any problems with material aging and consequently leakage.

For fuel flexible metallic hoses the supplier was tasked to carry out sampling and qualification testing.

A sequence of different testing will be performed, which comprises of

- visual inspection and dimensional check
- proof pressure and leakage tests
- x-ray and fluorescent penetrate examination
- pressure pulse test
- vibration test
- microscopic examination

The results of these tests are still outstanding.

Fuel Valves

During the Life Extension Program those valves were considered

- where high stresses are caused by medium pressure / hot fuel environment etc.
- where in-service experience had identified units to be safety critical and
- which are installed in areas where any leakage will cause a fire / explosion hazard. On-condition operation of these valves beyond their certified life of 4000 FH could cause additional safety hazards (fire / explosion), since the failure rate which has to be considered for these valves beyond their certified life is not known.

Each valve assembly consists of the valve and the actuator assembly. As far as the valve assemblies are concerned, the suitability for life extension is related to the valves only. Methods for extending the life of the valves beyond 4000 FH will be defined by the Supplier by means of Master Inspections performed with sample units. Part of this Master Inspection are the

- examination of all parts for corrosion, damage, wear and tear
- check of all rubber and elastomeric parts for hardness and quality
- examination of all critical parts with non-destructive and X-ray methods (where applicable)
- Direct comparison of the condition of the individual sample units.

In the event that the approved method of achieving the required 8000 FH Life Extension should entail Life Extension Qualification Testing, then this testing in general will comprise endurance and vibration testing

The Supplier has declared all actuator assemblies unsuitable for Life Extension due to their suspected

poor in-service MTBF and known design limitations. He proposed therefore to replace these actuator assemblies with superior equipment. These consist of a permanent magnet motor, which negates the use of the clutch assembly currently used on the Tornado actuators and offers improved switching characteristics with self-cleaning action. The new actuator will also offer better reliability than the existing one, consist of a hermetically sealed electrical connector and provides better EMC performance.

Fuel subsystem equipment

Many equipment from the fuel subsystem are part of life extension activities as they have been classified as safety relevant. These items are mainly different type of pumps, flexible sliding joints, thermal diffuser, etc.

Current results of the life extension process from sampling, analysis and re-qualification revealed that most of these equipment could be life extended. The DDPs will state the revised clearances and additional requirements for overhaul in some cases.

For the double-ended boost pump, the electrical harness, which is a safety critical component (high voltage power line which is submerged in fuel), shall be replaced after its original design life limits are reached.

Life extension of fuel coolers requires surge pressure measurements to establish in-service realistic conditions.

Fuel Tanks

The Tornado fuel tanks consist of bladder type cells of different construction. The tanks are not part of the life extension program as PANAVIA has already recommended to maintain the tanks in accordance with an "on-condition" lifting policy with no significant adverse effect on the flight safety of the aircraft, although reliability may be degraded.

Electrical Wiring

General

Electrical wiring is used all over the aircraft in all flight systems. Cable looms are distributed in and through many zones of the aircraft, where different environmental conditions exist as for instant heat, moisture, in case of leakage fuel, hydraulic oil, but also vibration, acoustic, etc. All these environmental effects need to be assessed for life extension of the electrical wiring.

The general design philosophy of the electrical system is as for other flight safety critical systems, that

no single failure does cause a hazardous situation. Therefore the wiring system is categorized as CAT3.

However wiring degradation with time is a significant problem as if not maintained it could cause a hazardous situation.

Wiring can fail in two ways:

- a) as an open circuit conductor and
- b) with a short circuit to another wire or frame

Case a) is not considered to be safety critical as flight systems are of fail-safe design. It is assumed that any functional failure within a flight system caused by the wire failure will be found by functional on aircraft tests.

Case b) is the much more significant failure as a short circuit to a metallic component or another wire can produce an electrical arc which in turn can cause a hazardous arcing condition.

The probability of this failure condition is considered to be quite small, as a series of correct conditions need to be present in the same time. But with increasing age of the wiring this probability will increase and thorough inspections need to be placed in order to slow down the increase.

Life extension philosophy

It has been evaluated that life extension of wiring by extended qualification activities may not lead to a life increase, as it could be shown that the consumed life is dependent on many factors which accelerate the aging of the wire insulation. Such factors are for instance the location, contamination by aggressive fluids, heat but also maintenance activities by which the wire looms are manipulated.

Therefore, the definition of a most cost-effective maintenance policy is the basic strategy for Tornado wiring life extension.

The following engineering activities are performed and methods investigated:

- Assessment of the suitability of the RAF WIDAS (Wire Insulation Deterioration Analysis System) as a sampling tool that provides for a preventative maintenance program of timely wire replacement.
- Zonal analysis (ZA), identifying the risk of wiring failure on a zone by zone basis considering the environment and other components installed in this zone. The main objective of these activities is to identify risk areas that would allow the definition of dedicated maintenance policies. These areas could then be subject to further investigation by other methods. The definition of risk areas would also remove the need of further investigation of low risk areas and so reduce costs.

- Infrared Thermography (IRT) for detection of wire failures or high-risk conditions will be investigated.
- Wire extended qualification testing in conjunction with the supplier. These tests consist of dry and wet arc and thermal life tests. Tests will be conducted by an independent laboratory and tests results send to the wire manufacturer for assessment.
- Investigation of other methods including high voltage (HV) testing.
- Review of other existing wiring policies within airlines, wire manufacturer, etc.

Electrical wiring survey

An investigation of the wiring condition in a large number of aircraft (49) has been performed in order to assess the state of wiring within different aircraft, which are operated in different air forces, at different environmental conditions, which are subject to different maintenance, etc.

It has been established that wiring deterioration is dependent mainly on the four topics:

- a) Applied maintenance policy:
 - Better quality of preventive maintenance (removing of oil leakage, dirt, ...) results in better condition of the wiring system.
 - The use of de-humidifiers has a significant, positive influence on the level of corrosion.
- b) Operational environment:
 - Aircraft operating under coastal or wet climatic conditions have shown more electrical and general corrosion.
 - As the aircraft spend only 2% of their total life flying, the conditions of the hangars are important regarding deterioration effects: warm and dry hangar conditions have positive effects compared to the aircraft standing outside.
 - Long duration flights at high altitude in dry air lead to a better state of the wiring system compared to for instance trainer aircraft, which perform much more cycles (take off and landing) and fly in lower height with much higher air humidity
- c) Materials used in the wiring system
 - Embrittlement and dry-out of tie wraps exacerbate chafing at tie wraps position, especially where tie wraps have been applied too tightly when fitted, which could be the result of unsuitable tooling.

- Use of heavy plastic spiral wrapping as a sleeving on flexible looms is unsuitable as encouraging insulation cracking at the point where the flexible part leaves the main loom.
- Perished rubber sleeves if they perish allow the wiring loom to open up, which could lead to increased chafing and rubbing
- Use of heat shrink material is not suitable as flexible looms become hard and can make wiring looms rigid.

d) Time in service:

- It has been established that contamination of the wiring system by dirt and oil can be related to the aircraft age but is also dependent on the frequency of cleaning activities

The findings of this survey will be used to establish recommendations for further usage of the wiring system for a limited period of time.

Flight Control System

General

The flight control system consists of various components, as for instance hydraulic and mechanical actuators, ball-screw actuators, electrical motors, computers, wiring, switches, control rods, lever, etc.. As for other systems, the effect of failures of any FCS component on flight safety hazards has been investigated taking into account the system hazards as defined in the safety analysis documents.

The major problems, which have been identified for many components, are related to fatigue and endurance. Failures due to wear and tear are the most reported service failures (Fig. 2).

Life extension philosophy

Although the FCS consists of a variety of different components, such as mechanical/hydraulic as well as electric/electronic components, life extension of these equipment will base on the following general work:

- Sampling of high aged ex-service units
- Analysis of the MTBF data.
- Theoretical analysis of life potential
- Re-qualification tests (endurance, fatigue)
- Investigation of aging effects on materials and components by sampling
- Establishing of overhaul, maintenance procedures

- Replacement

Mechanical, hydraulic components

The majority of safety relevant components have equipment DDPs, which define operational and other life limits.

Life extension of hydraulic components such as actuators will be based on sampling of high aged equipment, theoretical assessment of possible further life potential, re-qualification programs where possible and overhaul at defined intervals.

Life extension of mechanical components as for instance linkages, control rods, control stick rods, levers and rudder pedal mechanism are covered by the structure statement (chapter 2) and no fatigue life extension activities are necessary.

Flight Control Computers

The Flight Control Computers are designed and constructed according to MIL-Specs and safety requirements and consist of a triplex multi-channel computing system in two computer boxes (Lateral and Pitch), Fig. 7 (CSAS computer).

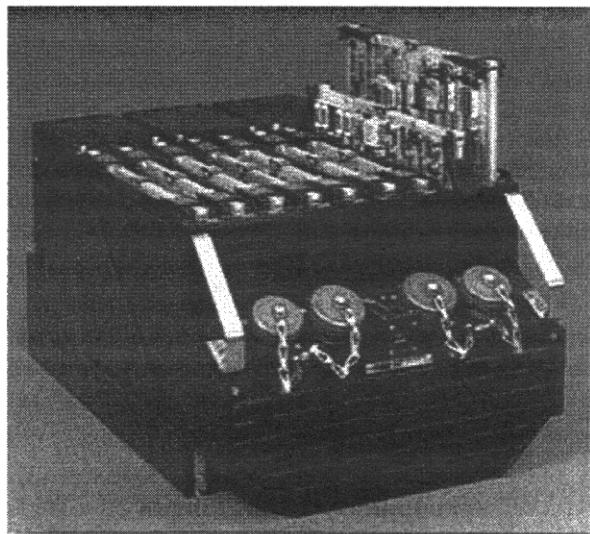


Fig 7: CSAS computer

The technology used in these computers is developed in the seventies and is therefore no longer state of the art. In-service support becomes more difficult the longer the computers will stay in service.

Within the life extension program the supplier have investigated various computer, which have been in service nearly 4000 FH. The general result of the survey is that only partially degradation of equipment was found which could be directly attributed to aging.

Some problems have been found which could be regarded to

- poor standard of repair and
- the application of conformal coating

but it is considered that these findings will not degrade the functional performance and also reliability.

Additional, minor damages were found, which could be caused by mishandling during maintenance work. It is therefore proposed to review the current handling practices so that no undesirable mechanical or electrical failures are introduced, which could have an influence on the performance.

The survey of aged electronic components has revealed additional problem areas as following described.

Aging problems with electronic components

- Relays

Relays are not pure electronic components but consist of many mechanical parts like springs, special contacts, coil and printed circuits and all parts are subject to continuous aging.

Relays are designed for a certain (high) number of operations and the degradation of the contact will result in an increase of contact resistance.

The relays which have been investigated in-depth were found in a good condition and replacement is only proposed if any failure is found or if they are physically damaged.

- Potentiometers

Variable resistors include mechanical and "Pertinax" parts which produce an electronic resistor by moving a mechanical part on the surface of a small "Pertinax" part.

During the aging process, environmental conditions and load could have an adverse effect on the performance of this type of resistor, which might result in a negative influence on the Build-In-Test-Equipment (BITE).

Investigation of aged potentiometers has revealed in some cases a degradation of the case seal and also the existence of hairline cracks, which would under certain conditions allow the ingress of moisture.

It is therefore proposed to investigate the general mechanical and functional condition of these components and replacement, if the component is defined as unserviceable.

- Capacitors

Capacitors of different style are used.

Aging problems have been experienced for instance with 'castanet' wet tantalum capacitors. The major findings are

- change in state of the liquid electrolyte, in which gas bubbles have been found
- loss of the electrolyte which results in corrosion around the anode/cathode seal, which results in a reduction of the capacitance value (μF).

- Printed Circuit Boards (PCB)

PCBs in old computers are partly wired. Furthermore, they carry connectors for the Dual Card Assemblies.

Problems have been experienced with surface leakage currents, which could be a dormant failure, contamination and condensates in conjunction with humidity.

Also local delamination of the top layer of the PCB was found in areas where solder pads have been repaired and probably excessive heat were introduced. It is therefore considered that the delamination is attributed to poor standard in-service rework.

Additionally, aging of the PCB's solder joints have been found, as shown in Fig. 8.

- Connectors

Similar problem as described for the PCBs have been experienced with connectors, which have connections to the PCBs and to the wiring via solder joints.

- Wiring (within Computers)

The major problems with wiring within the computers are cable chafing with nuisance effects.

- Resistors

It has been experienced that aging could change the tolerance of resistors. Therefore the logic circuit which the related resistor belongs to will change and this can produce failures and No-Go's during BITE.

- Transistors, Integrated Circuits (ICs), OP-amplifiers

Load-Life during different usage of the aircraft and maintenance may stress these electronic components.

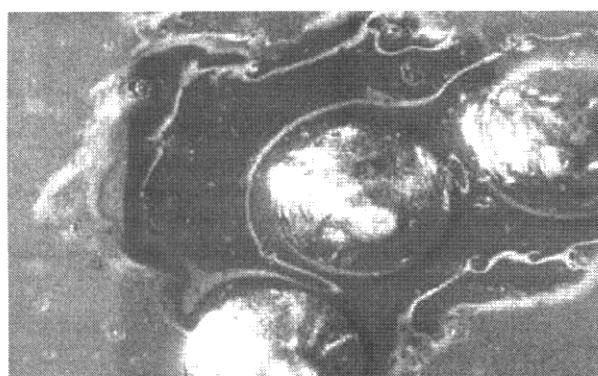
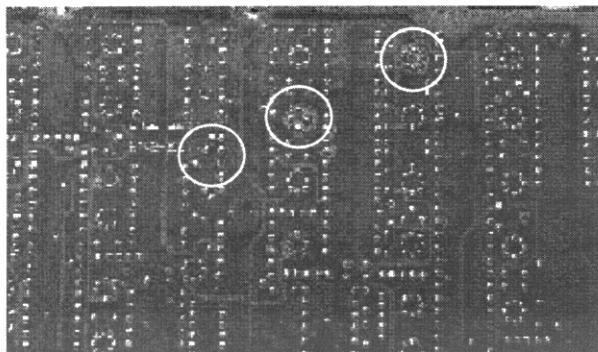


Fig. 8: Aged solder joints of PCB's

Influences of operational Environment

The Tornado aircraft is operated by different airforces in different environmental conditions. The Tornado is also used as a Navy variant. Therefore very different operational conditions exist, which have a different influence on aging.

The following should describe briefly the major environments which contribute to aging of electronic components of the Flight Control Computers and which influence is assessed during the life extension work:

- BITE-Runs (Preflight/First Line)
- High altitude flights
- Power ON/OFF cycles

- Different Weather conditions such as high/low temperature, high humidity, lightening, salty atmosphere

An example for corrosion due to environmental influences is the corrosion around power resistor leads found due to Doghouse - environmental influence, Fig. 9.

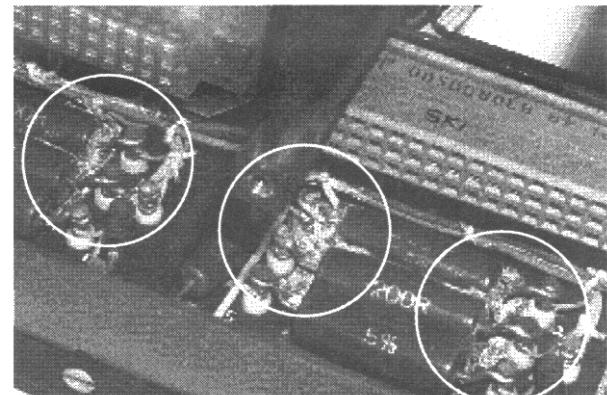


Fig. 9: Corrosion around power resistor leads

Additional, the following influences could have a degrading effect on component integrity

- Operation in Navy / Air Force

The Tornado is designed for different combat roles. The equipment, which is operated within Navy aircraft, is exposed to salty atmosphere and therefore the effect of corrosion becomes more critical for these types of aircraft as they are often flying over ocean.

- Wars

Wars like the past Gulf and Bosnia will have special influence as normal flying in squadrons from home base. Heavy stores and more agile flying will stress not only the airframe but also the electronic equipment (cards, boxes, wiring, connectors).

- Parking

Normal Parking of A/C in a shelter with covers on all potential sensors like the ADD-Probe is needed for a long life. Parking on and outside airfield and sensors not covered is a potential hazard for fast aging of electronic equipment and their sensoric system.

- Storage

Storage of electronic equipment is not the same as for mechanical parts. The electronic items will age under these conditions as well as under normal usage.

To investigate the life consumption of electronic parts, the correct service history need to be assessed. Therefore it is essential that configuration control is possible for all critical equipment and components.

Life extension of FC computer by overhaul

As for other equipment, the supplier will give a guarantee for his equipment only for usage within the specified operational and life limits.

Extended qualification of the Flight Control Computers was discussed to fulfill the customer requirements for life extension. But due to the high costs of this re-qualification process a different approach was examined with the supplier customer.

It was decided that life extension of the FC computer will base on overhaul after the equipment has reached its certified service life.

The procedure base on knowledge of the supplier maintenance work and by sampling of high aged service equipment:

- Survey of early manufactured computers to identify time dependent effects, which will be used to produce test and inspection procedures
- Research failure and repair records required from service
- In depth investigations of
 - PCBs
 - Connections
 - Active components (i.e transistors)
 - Passive Components (i.e capacitors)
 - Case structure

These investigations will be used to establish overhaul procedures for the different type of computers. These procedures are currently under strong discussion with the customer and the suppliers and the following general measures are foreseen:

- Detail examination of the physical condition, identifying of aging problems and components which need to be replaced
- functional check
- cleaning
- EMC Test: Early standards of the Flight Control Computers are EMI hardened. Therefore an EMC test will be conducted after the overhaul procedure has been successfully performed.
- Conducting of an Automatic Test Procedure (ATP)
- Release to service

Obsolescence of electronic components

Review of the availability of electronic components for the FCS computer revealed that the obsolescence problem becomes more and more critical.

To overcome this situation the following options are in discussion:

- a) Lifetime buy of components, which will be no more available in the future. Presently up to 50 electronic components are no more available on the market. This concerns especially FET switches, Power Transistors, PROMs, Optic couplers, special A/D converter, TTL components, Resistor - Thickfilm networks, etc.
- b) Redesign using actual standards of components and modern production processes to achieve compatible equipments on LRU level.
- c) Development of new concepts, new technology with new hardware as well as software, together with limited measures related to alternatives a) and b) for the interim period of years until completion of introduction in service.

Presently the way forward is not yet decided and will base on the experience and requirements established within the life extension program

Hydraulic System

General

Safety relevant hydraulic subsystem equipment and components have been identified based on assessment of the *System Safety Assessments Report*.

The major subsystem failures modes are

- Loss of hydraulic power and hydraulic flow which would lead to loss of aircraft control
- Loss of emergency equipment function required in the event of an emergency

The evaluation of age related system failures, Fig. 2, have shown that leakage is the most significant reported in-service occurrence, which is well understood, as endurance of hydraulically functional equipment will lead over time to a degradation of the seals. This means that the older the equipment stays in service the often unscheduled maintenance will be required to address these problems.

Life extension philosophy

The life extension philosophy for the hydraulic subsystem equipment and components base mainly on sampling and re-qualification of equipment and the definition of maintenance and overhaul requirements.

A Master Inspection for investigation of the condition of hydraulic subsystem components in aged aircraft is currently not performed.

Hydraulic Subsystem equipment

Many equipment and components have been selected for life extension as for instance pumps, pressure switches, accumulator, valves, self sealing couplings, reservoir, slide and swivel joints, etc.

Aged equipment from service have been delivered to the suppliers for detail investigation of the condition after 4000 FH in service and to state the serviceability for further usage.

Most of the hydraulic equipment will be re-qualified by extended impulse-fatigue tests, endurance and fatigue tests.

Current results indicate that most of the equipment, which are subject to life extension, will be suitable for 8000 FH. An overhaul after 4000 FH will probably be required by the supplier after the original specified life is reached.

During this overhaul the units will be resealed and areas, which have shown wear and tear needs to be repaired, if necessary.

Aging of equipment (corrosion, wear and tear) as well as contamination of e.g. filters need to be addressed by appropriate in-service maintenance procedures.

Hydraulic piping

Fatigue life of hydraulic pipes is not of concern. Therefore no activities for life extension are foreseen.

Aging effects, as corrosion of the high-pressure steel pipes need to be addressed by appropriate maintenance activities. Nevertheless, leakage is not considered as safety critical and severe leakage will be detected early by pre-flight or post flight checks.

Hydraulic hoses

Life extension of hydraulic hoses is in discussion and there will be different solutions for high and low pressure hoses. It is likely that high-pressure hoses need to be replaced after normal service life.

Significant aging effects on high aged hoses have not been experienced.

Environment Control System

General

The ECS subsystem consists of a number of sub-subsystems as for instance cabin pressurization, engine anti-icing system, temperature control system, windscreen heating system, etc.

Again, based on the original *System Safety Assessments*, safety relevant equipment have been identified and the customer has defined equipment for which life extension should be carried out.

Mainly all equipment and components, which belongs to the high pressure and high temperature part of the ECS system and subject to engine bleed air conditions are critical.

The components which belongs to the low pressure ECS part are categorized as non safety relevant. No specific life extension measure are foreseen and also no investigations into aging problems are performed.

Life extension philosophy

Life extension measures for ECS equipment and components are basically:

- Sampling of aged equipment
- Stress, fatigue analysis
- Definition and conduction of re-qualification tests
- Master inspection for critical zones of intercooler ejector line
- Definition of maintenance procedures

ECS Subsystem equipment

Equipment, which is subject to life extension activities, are for instance valves, pre-cooler, reservoir bottle.

These items are subject to life extension measures as defined before. The investigation of aging effects on equipment is performed during sampling of high aged components.

ECS ducting system

A Master Inspection is proposed, which should investigate the critical zone of the intercooler ejector line.

The main objectives of the Master Inspection are

- Investigation of installation aspects to support of life extension of ducts and critical bellows including investigation of bellows inner condition to enable decision for re-qualification testing of bellows / structural interface
- Investigation of the physical condition related to aging aspects, mainly external corrosion beneath the insulation

The investigations have not yet started.

Part 4

Conclusion

The Tornado life extension program was defined to extend the life of the aircraft up to 8000 FH. Within this program, which is still in progress, a clear process has been defined for subsystem life extension, which focuses mainly on safety relevant equipment and components.

Part of this process are investigations into the aging condition of equipment and components, which are near to the end of their certified life. The results of these investigations, mainly got by sampling of aged equipment, Master and Zonal Inspections, are used to support the definition of requirements for further extended use of equipment beyond their certified life.

Aging problems, such as corrosion, wear and tear, leakage, or degradation of electronic components, etc. have been identified in many subsystems but fortunately none of them required immediate action to prevent a safety critical hazard. Aging problems will be addressed by overhaul, repair or replacement actions, dependent on the condition of the item investigated, but also on factors like economics, spares availability and overall fleet management.

Experiences we made during the time carrying out the life extension program:

- Supplier are mostly unaware of the in-service experience with their equipment
- Strong and time consuming discussions were necessary between customer, SDR and supplier to define a by all involved parties agreed approach for life extension for certain equipment
- Service history and life consumption cannot be provided for many equipment, which means that the point where the equipment should be subject to agreed life extension measures (i.e. overhaul, replacement of the complete unit or detail parts) is difficult or even not possible. Recording equipment life would help minimising operating costs as the full performance of the equipment can be used.
- Non availability of service documentation resulted in the fact that provision of suitable loan equipment for investigation at the supplier was difficult to locate.
- The effort undertaken to re-qualify equipment beyond their original certified life is often significant. Since in the Tornado program most equipment have in the past only been qualified up to their original specification requirements it should be considered much more cost effective, if

the equipment would have been tested to failure ideally just after normal qualification to spec has been completed.

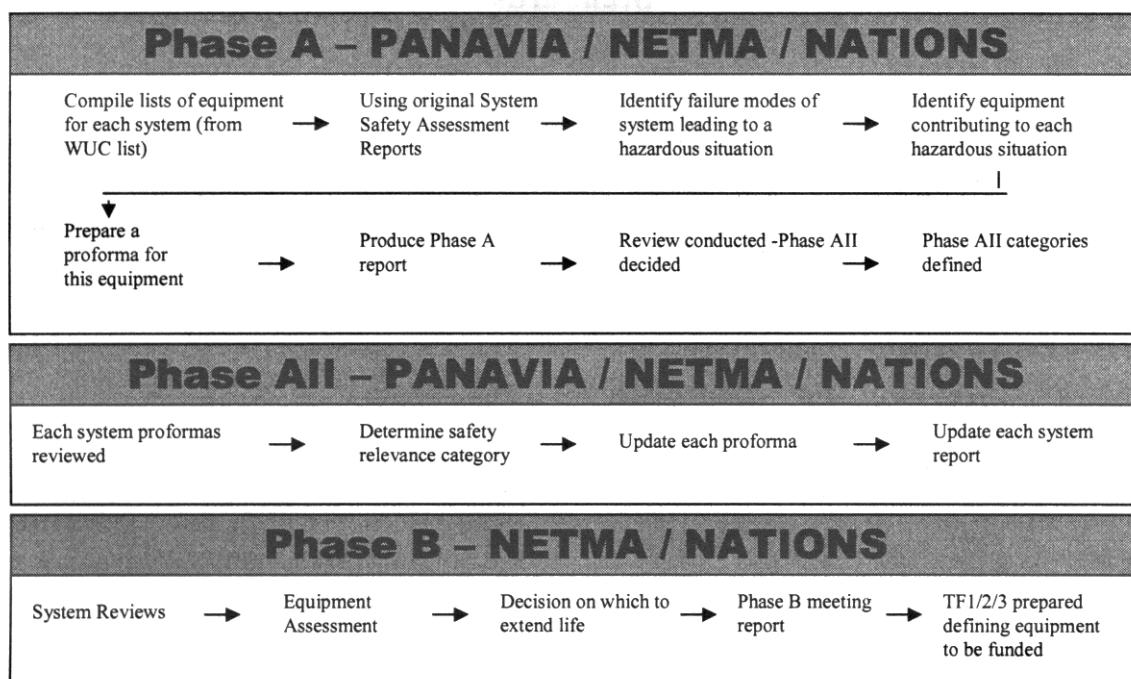
As general conclusion it can be stated that life extension of Tornado subsystems is successful as in many cases the customer requirements can be met.

PANAVIA and their supplier have or will clearly specify overhaul, inspection and maintenance requirements for equipments which failure would be hazardous and which should be mandatory for further safe operation. Aging aspects will be covered by these inspection requirements.

References:

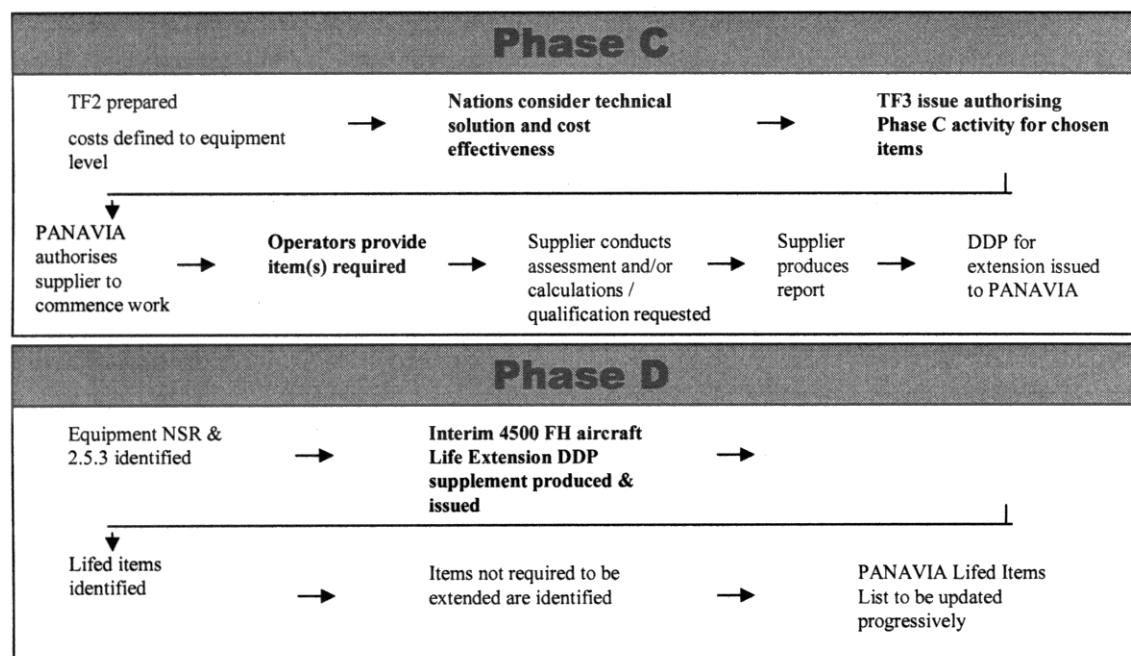
1. PDT 1299: TORNADO Life Extension Process Definition Document
2. PDT 1299 Phase A Report
3. Various reports and presentations of inspection results

PDT 1299: Life Extension Phase A/B

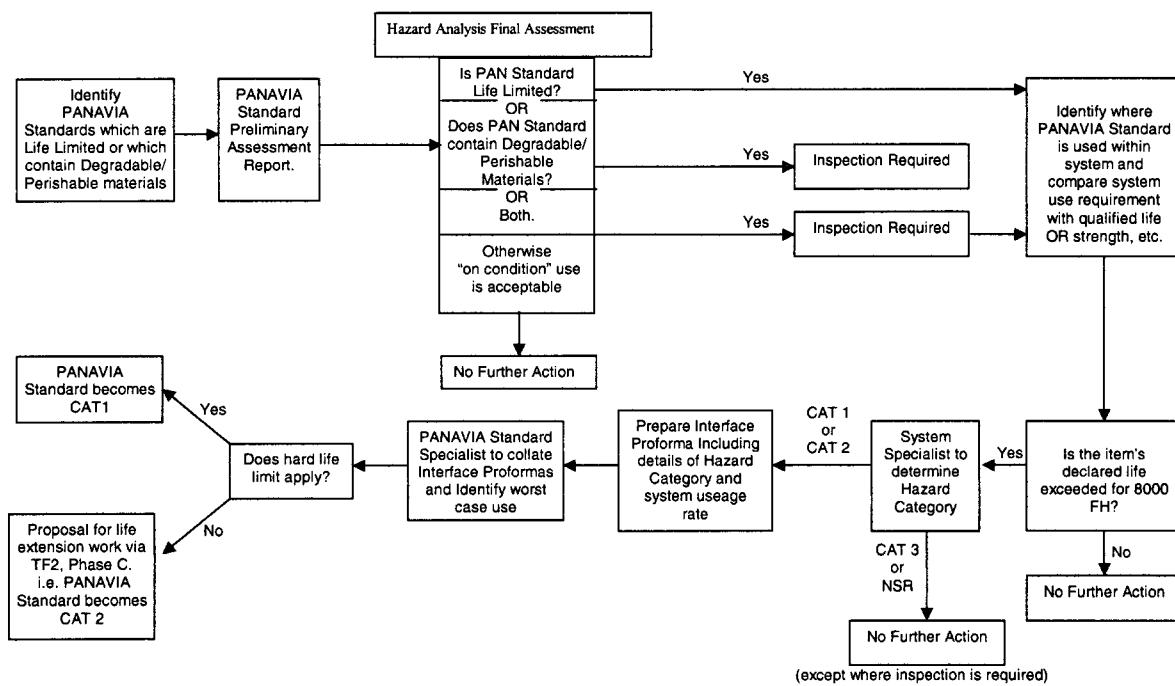


Life Extension Phase C & D for Flight Systems & Equipment

(Items shown in bold letters indicate customer / operator dependency)



Process for identification of Life limited, Safety Relevant PANAVIA Standards



SAFETY AND SERVICE DIFFICULTY REPORTING

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Today, safety is considered to be of highest importance in most societies. In the context of the military, safety is essential to averting loss of life and damage to a high-value asset. While safety may take second place to winning a war, its importance is further accentuated because of its connotation to battlefield readiness. There have been numerous instances to illustrate this last point. To wit:

- Widespread Fatigue Damage (WFD) was discovered in “weep holes” of fuel tanks of some C-141 military transport airplanes. Because of the loss of minimum residual strength, with the attendant risk of catastrophic fracture posed by WFD, the entire fleet had to be grounded and an expensive refurbishment program had to be undertaken before the fleet was deemed to be airworthy. In this instance, the unsafe condition was detected and corrected quickly, so no lives were lost nor did any of the airplanes in the fleet suffer catastrophic damage. However, the grounded aircraft were certainly not battle-ready for a certain length of time. Had they been sent into battle, they would have had to be operated under severe flight restrictions and, thus, their utility to serve the purpose of the deployed forces would have been very restricted. Had they been deployed without any restrictions, in all probability they would have been unable to complete their missions and the Air Force could have lost valuable aircraft assets. Also, the necessary logistic support to properly carry out tactical operations in the battlefield would not have been available.
- WFD was the primary cause of a highly publicized air accident involving a commercial aircraft. The wide publicity given to that single accident, abetted by on-site video tape recording of the condition of the aircraft after it had landed, shook the confidence of the public in the safety of commercial aviation. As a result, inspection and refurbishment of 3000 jet transport airplanes among a fleet of about 5000 was mandated by the authorities, to be undertaken on an urgent basis. The economic impact of this mandate on the airlines, the aircraft manufacturer and the flying public was high and resulted in numerous complaints to the regulatory authorities. It must be noted that since that time more than twelve years have elapsed without a single accident attributable to WFD.

These instances explain my motivation for including the subject of safety during this Lecture Series. However,

the subject is extensive and so many books have appeared that address some aspect or the other that my remarks are meant to complement the existing literature. Much of what I intend to share with you today is not something I have developed on my own, rather it has been influenced by my comrades and peers when I was in the civil aviation community.

Scope of the Lecture - Analysis and Data Requirements for Assessment of Operational Safety:

An aircraft is an assemblage of complex and highly integrated sub-systems - the structure, the power-plant, the electrical, the mechanical, and hydraulic systems, the avionics suite, the human-in-the-loop to name a few. To eliminate the risk of the sub-systems to fail, individually or in concert, safety analyses are routinely performed by aircraft manufacturers. The manufacturer also conducts analyses to ascertain the consequence of a failed part to assure that it does not in any way threaten the safety of the entire system.

Before an aircraft model enters service, whether for military or civil use, the design has to satisfy a rigorous set of requirements, which are governed by regulations. These requirements include an analysis of the probability of failure of each component and the hazard caused by the failure. This subject, termed as “Systemic Safety [1],” will be beyond the scope of this lecture. Rather, the remarks will concentrate on the operational phase of the aircraft’s life. That is the phase subsequent to the aircraft put into operational use for the first time.

However, keep in mind that before the aircraft enters the fleet, there are numerous design reviews, ground and flight tests, and production approvals that are required to assure that the aircraft is safe and able to perform as intended in the operating environment. At times, the origin of problems that are encountered in service may be inherent in the design or the manufacturing stage or due to construction methods. For instance, an element in the chain that led to the failure of the commercial aircraft mentioned earlier was a failed bond. The failed bond resulted from an inadequate bonding process. It created stress risers at the rivets, which were designed to merely serve as secondary conduits for transferring load. The resulting fatigue cracks were aggravated by loss of material due to corrosion, resulting in intrusion of moisture from condensation and precipitation. Such problems that are encountered in service must be quickly corrected in order to prevent accidents and to maintain battle-readiness of the fleet. An essential requirement for quick resolution of these type of

problems is a technical team that is familiar with not only the design features of the aircraft model and any subsequent modifications that had been effected previously but also the original design philosophy that guided the design. Often, it is beneficial to retain some members of the original design team to serve in the maintenance group in order to maintain the necessary know-how.

Measurement of Safety

In order to assess safety of a system after it enters service one must define safety and establish a set of metrics (measurement standards) for safety. A metric may be the number of failures per one thousand operations, or it may be an incident rate or an accident rate. Such gross metrics are normally refined by dividing the accidents into categories by causal relationships. Furthermore, metrics are often normalized in terms of usage. In any event, the establishment of safety metrics has been subjective, to say the least, and a bit disorganized from the standpoint of relating the accident cause, the events leading up to the accident, and the design fix. The problem is best illustrated through Figure 1, and 2. Both figures have been extracted from publicly released Boeing Airplane Company documents [2, 3]. They depict the relative risk of an accident as a function of the phase of flight, based on historical data. Clearly, if miles flown is chosen as the normalizing factor for a safety metric, the metric chosen ignores the fact that risks between destinations involving multiple flight legs and the risk involved for a single leg, for the same distance traveled, are unequal - hence, the metric would be inappropriate. Similarly, in the assessment of military aircraft, the hours of operation is usually chosen as the normalizing factor but such a choice ignores the fact that the mission profiles could be vastly different, even for the same aircraft model but used in different squadrons. Thus, the establishment of multiple metrics for risk using the same database increases the opportunity for establishing a correlation between data and risk, thereby making the safety management system more robust.

Accidents and the Role of Precursors:

It is generally agreed that there exist certain precursors to each accident and incident. If one of these precursors is not recognized and the underlying condition that has caused it is not corrected in time, then it can graduate into an incident or even an accident. Aircraft are highly engineered systems, endowed with redundancies and fail-safe features. They are "noisy" systems. That is, they can give so many indications, of which only a few are precursors, that one can easily be lulled into complacency. Fail safety embraces two concepts. One is the concept that the first failure does not impair functionality of the system. The second is that the first failure must be obvious to the extent that it will, in all likelihood, be detected well before the onset of subsequent failures, which may endanger the safety of the system. Thus, the first occurrence of a service

difficulty associated with a sub-system in an aircraft is a prospective precursor of progressive failures that could result in an incident or accident. Furthermore, multiple occurrences of service difficulties, especially after corrective actions have been attempted, are indicators that the risk of an incident or accident is rising. To take full advantage of being given such warnings, the organization responsible for safe operation of the aircraft must systematically collect reports of service difficulties. Just as importantly, this same organization must systematically and expeditiously analyze the reports being collected to establish their root cause of the difficulty or difficulties and its potential for a resulting accident or incident. The analysis must be accomplished early in order to allow sufficient lead-time for corrective action to be taken. Even with a service difficulty collection and analysis system in place, the organization will be unable to use it to reduce or eliminate incidents and accidents unless higher management in the organization recognizes their value and directs development and implementation of corrective action. Clearly, improved safety will result if attention is more focused on precursors.

Detection of Service Difficulty

A Service Difficulty is symptomatically manifested by one of the following:

Visual, such as cracks, warning lights, observation of smoke, etc.

Aural, such as alarms, abnormal sounds, etc.

Tactile, such as excessive vibration, electrical shock, stick response, etc.

Olfactory, such as fumes from electrical systems or oil or rubber, etc.

Response to transducer devices such as those used for nondestructive inspection of structural components.

Service difficulties can manifest themselves during airworthiness inspections and other maintenance related activities. One example is the detection of a structural fatigue crack in an area adjacent to the area being inspected. The maintenance program had no instructions for inspecting this cracked area. Had the service difficulty report not been filed on this crack, and had a single observant authority representative not discovered this difficulty report and investigated it, further crack growth in this area and other aircraft might have occurred and graduated into something serious.

It would be erroneous, however, to draw a correlation between the number of service difficulty reports generated and risk. A large number of reports may mean that the operational and maintenance personnel are alert and diligent in reporting discrepancies, not necessarily that the risk of failure is rising. In this case, it may

simply be a tribute to the robustness of the inspection and maintenance program. Only systematic analysis performed by trained and knowledgeable analysts can correlate the risk level to the number of service difficulty reporting rates.

Analysis and Data Requirements

There is a symbiotic relationship between: (a) the purpose of safety analysis, (b) the methodology to be used for evaluating safety (or risk), (c) the data required to perform safety analysis, (d) the confidence to be reposed in the results, (e) and the burden of the data collection effort. All five aspects will have to be considered in concert to devise a robust system that balances system costs (figure 3).

Safety analysis may be required for a variety of purposes. For instance, to gage the general health or safety of the fleet would require a different methodology and could be accomplished with an abbreviated set of data elements than what might be needed for a forensic analysis of an accident or incident. Thus, the circulation of a questionnaire among the various groups involved in maintaining safety to establish the connections between analysis methodologies that are being used or desired, and the respective data requirements is advocated.

Aircraft systems are becoming more and more complex, placing more sophisticated demands on data collection and analysis methods. Also, the increased attention being given to safety and the accompanying demand for data driven safety programs, makes the data elements that would have been considered adequate in the past appear as lacking in precision and detail. Thus, the number of data elements, the extent of detail to be included in any gathering effort, and the configuration of the database itself should be designed to allow for some growth in data requirements. It is imperative that an organization designing a service difficulty reporting system that mandates the collection of certain data elements simultaneously considers the analysis to be conducted of the collected data. Many existing databases, such as the Service Difficulty Reports being maintained by the Federal Aviation Administration have come in for criticism [4]. These databases collect many pieces of data that are not used or are redundant. Such databases are primarily designed to facilitate the collection of data but with little or no attention being paid to the needs of the analyst to correlate the data with the airworthiness of the individual aircraft or the fleet. Hence, it is advocated that a safety program - any safety program - be revisited, perhaps re-tuned, every five years, both from the viewpoint of currency and adequacy.

Avionics-related malfunctions may have serious implications in terms of safety of new generation aircraft. These systems are being given more authority over primary flight control of the aircraft. Thus, the reporting of associated malfunctions, defects, and failures become more critical to proactive safety

analysis. Their failures during any phase of operation may have safety implications. In any event, data should be collected to support explicit program requirements. Terminology such as "abnormal or emergency actions" and "endanger the safe operation" in regulations will not provide consistent reporting without further definition and guidance.

The distinction between reliability and safety is much debated in the context of data requirements. It has been argued the data needed for performing safety analysis is not as extensive as that for maintaining reliability. However, with the emergence of the nearly synonymous philosophies of Reliability-Based Maintenance and Condition-Based Maintenance, which takes the risk of failure(s) into account, the distinction is blurring.

Hand-held electronic devices have eliminated much of the paperwork in data gathering. Such devices make possible the gathering of voluminous data without making the data gathering effort either burdensome or time consuming. In fact, the development of software that can readily depict on a hand-held device the geometrical layout of components as well as the inter-connectivity of the functional units would make facilitate acquisition of data that capture more details about a malfunction or a failure than is now the case. Electronic entry of data has another great advantage, viz., it avoids data corruption due to transcription errors and expedites the addition of more data elements to the database.

Data Standards

The term "data quality" can at once mean different things, such as erroneous data, inconsistencies in the data, insufficient detail that has been captured in the data, completeness of the data sets, etc. Each of the meanings has a bearing on safety. For instance, there is a wealth of data about instances of cracking in airframe structures but they are not very useful because of lack of precision and standardization. From the standpoint of systematic analysis of large quantities of data, the most important attribute of a safety related database is consistent reporting. The adoption of a common terminology is one aspect of consistency. Clarity of terminology is a related aspect. A critical need for data that is stored in relational databases is that fields should be assigned in each data record (report) to allow for supplementary comments by the mechanic. The FAA maintains one of the largest safety database in the world, the Service Difficulty Reporting (SDR) System. However, because the SDR is a relational database, no provision has been made for supplementary notes. For instance, the database does not allow the mechanic to record the specific location of a crack, even if one is found in a principal structural element. As a result, many users rely on the SDR system only to confirm critical problems that have already been found or suspected - not to give precursory evidence of potential incidents or accidents.

Table 1 exemplifies a form for data recording, which would make possible supplementary notes to be made by the mechanic or inspector. The form for reporting incidents was devised by an internal FAA team, of which the author was a member. The data requirements for reporting service difficulty can be developed in an analogous fashion.

In the military context, harmonization of data standards with our NATO allies will inevitably result in more robust safety systems for all concerned. Also, since the occurrences of many types of malfunctions are rare, harmonization will allow data to be shared between nations that operate similar aircraft systems and increase the data pool, thereby decreasing uncertainty inherent in statistics-based analysis schemes.

Completeness of data, whether the entry relates to deviation, malfunction, or wear is nearly as important. The need to report and record every deviation from the norm, even though the vast majority of cases are benign, cannot be over-emphasized. It is also essential for the analyst (or analysis group) to promptly acknowledge receipt of each report and, once the analysis of a report is complete, to communicate the results to the maintenance group. Otherwise, the latter group may lose faith in the system.

Data Archival and Retrieval

An efficient database storage system has to take into account several factors. Simultaneous access to multiple users may be one requirement. Inclusion of pictures, and documents in the database may be another. There are several ways to store and present data and several types of database management systems (DBMS) have been devised and are commercially available. In choosing the right type of DBMS it is important to consider the capability of a typical user and the purpose underlying the use of the data. For safety analyses purposes, the DBMS should be capable of storing and manipulating complex objects and data types efficiently. The most suitable type and currently available DBMS are the ones known as object-oriented DBMS. Such relational databases allow for computer-aided searches and sorts that are simple to implement, allowing the user to concentrate on deriving the information he or she is seeking rather than focusing on the design of the database extraction tool. On the other hand, if one is willing to invest in more complex search engines, the database may need to be less structured and therefore contain much more information. An explanation of the various types of DBMS can be found in reference [5]. Even object-oriented DBMS have their drawbacks and, thus, the entire subject deserves research attention.

Analysis Methods

Service difficulty data can be used for a variety of purposes and in a variety of ways. The common thread that runs through all of them, however, is risk mitigation. Obviously, the criticality of the component

associated with the data, the number of incidences of failure, the consequences of failure, the method(s) used for analysis, the confidence band inherent in the analysis results, and the statistical character of the occurrence are inextricably related.

Accidents and, to a lesser extent, incidents and malfunctions typically involve a chain of events. The chain may simultaneously involve a design deficiency, a defect induced during the manufacturing process, improper maintenance or other human factors. Some aspects that are frequently involved are given in Table 2.

It has been argued that, since many factors are involved in causing an incident or accident, the safety management system should be highly centralized. The author would argue in favor of the opposite, mainly because the safety system would be redundant and, hence, more robust. The responsibility for safety should be divided into sub-groups, whose prime responsibilities are related to maintenance or air traffic control or some other factor identified in the table. Each group should be persuaded to believe that they are ultimately responsible for safety and each group should be allowed to devise their own system for monitoring risk. Of course, each such group will be much better versed in their own specialty and might tend to give greater attention to it. On the other hand, it can be argued that they will tend to take less for granted in other specialty areas and therefore subject them to greater scrutiny.

If the aforementioned view is accepted, it would follow that each group will have different data requirements. The latter can be fulfilled with relative ease by customizing data, but which is drawn from the same master data pool.

One example of an extensive and well-disciplined service difficulty reporting and collection system, as has been previously mentioned, is that being maintained by the FAA. Unfortunately, the FAA does not have the means to systematically analyze the data reported, which purportedly is not all-inclusive. Instead, it does so in an ad-hoc manner. That is, it researches the database to seek service difficulties that indicate the pervasiveness of a fault in the aircraft fleet. Such searches are carried out after the problem has been brought to the attention of the authority through other means, such as an incident or an accident. However, the efforts of the FAA are a valuable adjunct to the safety analysis efforts by industry. Moreover, the SDR database is accessible to other users, such as aircraft manufacturers and operators, who, because of their focus tend to be more systematic in the analysis of the data.

Causal Analysis

Causal analysis of an accident or incident seeks to establish those factors that were judged to be directly responsible in causing the event (primary causal factors) and those that contributed to the event (secondary causal

factors) by deconstructing the accident. For these causal factors, a causal chain can usually be established for each accident or incident [6]. The advantage of causal chain analysis is that in the case of multiple causes and multiple accidents or incidents, the common events or elements in the chain can be identified and subjected to greatest attention. Thus, the safety system can concentrate on those common events and maximize its responsiveness and effectiveness in cutting down-times, and reducing or eliminating accidents. The perceived disadvantage of this approach is that it is reactive rather than proactive. That is, the regulating authority and the industry (or the military operators) seek to eliminate the causal factor after the accident in order to prevent accidents due to the same cause from happening again.

Causal analysis does have an advantage over simulation and technical conjecture in that it is based on factual data rather than models that mimic a hypothetical event or engineering judgement, which relies on the knowledge base and experience of the technical team. Moreover, as has already been mentioned, in today's aviation industry, it is difficult to retain an engineering team that is intimately familiar with the continuous changes in the aircraft design after production begins.

The causal analysis approach, however, also suffers from the disadvantage that the analysis has a good measure of subjectivity, both in regard to the list of factors and their relative contributions. Also, due to the inter-dependencies of the various factors, such as those listed in Table 2, that are frequently encountered, the relative weights ascribed to the various causal factors can vary a great deal, as a function of the analyst. Thus, an intimate knowledge of the aircraft system is a prerequisite for someone engaging in causal analysis. The challenge of managing aircraft safety is identify and focus on truly hazardous conditions, so they can be eliminated before a potential accident becomes a reality.

Trend Analysis

One simple and effective method is used in the Aviation Safety for Accident Prevention (ASAP) program that is used by the FAA's Rotorcraft Directorate in Ft. Worth, Texas. The program selects components that fail by part numbers. For each part, it reviews the service history for 3, 6, 12 or 24 months periods. Based on the counts of service difficulty reports involving the part number, it predicts trends.

A risk level is assigned to each report. ASAP has the ability to quickly research whether an accident had a service difficulty history. For example, responding to a fatal accident involving the tail rotor driveshaft, the analyst was able to track part numbers, and identify five service difficulty reports that had found the part to have been worn beyond limits, and contained cracks or corrosion. Two of the reports described the results of inspection to be a sheared tail rotor driveshaft. Based on the accident and the supporting trend indicated by the

service history, the Authority issued an Airworthiness Directive (AD). A year after the issuance of the AD there were no more service difficulty reports, citing that particular part was reported. But, more importantly, the incidence of sheared rotor drive shafts has been drastically reduced. However, ASAP has one drawback: usage of ASAP is not yet proactive in that the analyst must be prompted by an event, such as an accident or incident to conduct trend analysis on a given part or component.

Monitoring of Safety Through Performance Indicators

The FAA's Flight Standards Service has developed a heuristic-based system called Safety Performance Analysis System (SPAS), primarily for the benefit of their corps of safety inspectors. They started building the system by getting teams of highly experienced and proficient inspectors together, with each inspector identifying the parameters that he or she uses during surveillance of an operator or a repair station facility. Each team discussed each of the identified parameters and developed a consensus about the relative importance of the parameters that must be scrutinized. Next, the parameters were weighted according to their perceived importance and aggregated into groups, with each group being termed as an "indicator." The advantage of a system that is based on indicators is that pools the knowledge and experience of the "gray beards" or the more experienced inspectors in the regulating Authority for use by the younger, less-experienced inspectors. Hence, it focuses attention on what is a warning rather than on events that are merely "noises." The disadvantage is that a rational derivation of threshold values, which signal caution or even danger, is not possible.

A variation of the idea of performance indicators as measures of safety is proposed by the author. It is based on "wiring diagrams" of sub-systems being used in conjunction with the concept of indicators. In the pristine condition, every cell in the wiring diagram would be colored white. When a failure of a certain part occurs, the analyst assesses the criticality of the part to flight safety and assigns a hue to that part (cell) in the wiring diagram. A deeper hue or color would signify that the part has a relatively high criticality. The wiring diagram is constantly updated by adding more color to the particular part to reflect arrival of new service difficulty reports. Two events will attract the attention of the analyst. The first is the depth of the hue of a certain cell and the second is the contiguity of cells (the ones that are sequentially tied or represent the redundant feature), in terms of their function, that are hued. The idea is based on the recognition of the fact that in both cases the risk of sub-system failure is increasing, and that the wiring diagram pictorially represents the rise. In fact, it would be relatively easy to convert the logic into a computer code that automatically raises a flag in either case, which cannot escape the attention of the analyst. Also, different colored flags may be set up to indicate

the level of alert. The scheme will also need to take into account replacement or re-design of the part, or the sub-assembly itself. That is also easily done by washing out the color in the particular cell representing the part or in the block of cells if the sub-assembly has been redesigned or refurbished.

Probabilistic Risk Analysis

Several probabilistic approaches to safety have been proposed [7]. However, such approaches are not looked upon with enthusiasm because no one wants to look upon safety management in a manner that resembles a game of chance. However, there are at least two major advantages of a probabilistic approach. First, it takes into account the variability in the data as well as the trends in the number of occurrences. It also provides for considering the relationship between seemingly unrelated occurrences. The analyst must examine the estimated probability of an accident, given a high probability of the occurrence of service events, and determine if intervention is required. A unique advantage of the probabilistic approach over a deterministic approach is that it enables the Authority or the SafetyOffice in the military to focus on the most likely causes of hypothetical, future accidents, and prevent them. By far the most important advantage is that it enables the Authority, and the operators, to get ahead of the power curve - that is, to correct the condition before the first accident occurs.

Concluding Remarks

As new technology is inducted, aircraft systems will inevitably become more complex. New technology generally means better performance and lower costs but there might be safety-related challenges as well. Also, increased usage and operating missions beyond what was envisaged in the design stage will magnify the accident rate as well as the fatalities, injuries, or losses of high-value assets. Safety systems will need to be more sophisticated and better methods of analysis will need to be employed. Authorities, and in the case of the military - themselves, will need to focus more on preventing accidents due to service related events rather than using service data to confirm the analysis of accidents that have already happened.

Concomitantly, more extensive data requirements and data archival systems will need to be engineered. Thus, the cost of maintaining a high level of safety is bound to rise but the cost due to not having an effective system will be many times greater. Safety of highly engineered systems, like aircraft, has a high price tag but the alternative will prove to be much, much more expensive.

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TABLE 1: EXAMPLE OF A FORM FOR RECORDING AN INCIDENT

BATCH # _____	I.D. # _____	
<u>REV.</u> <u>DATE</u>	<u>ANALYST</u>	<u>REVIEWER</u>
0 ____/____/____	_____	_____
1 ____/____/____	_____	_____
2 ____/____/____	_____	_____
3 ____/____/____	_____	_____
<u>EVENT ID NUMBER:</u> ____/____/____ YY MM DD SE	<u>TIME OF EVENT:</u> (SELECT ONE) UNKNOWN _____ UT _____ LOCAL TIME _____	
<u>EVENT CLASSIFICATION:</u> HAZARDOUS _____ MAJOR _____ MINOR _____ DAMAGE _____	<u>LOCATION:</u> DEPARTURE AIRPORT _____ DESTINATION AIRPORT _____ EVENT LOC. (CITY) _____ COUNTRY (EVENT) _____ LAT/LONG _____ UNKNOWN _____	
<u>AIRCRAFT:</u> TYPE-SERIES _____ A/C MAKE _____ FUSELAGE NO. _____ DATE MANUFACTURED _____ TAIL NUMBER _____ SERIAL NUMBER _____ ENGINE MAKE _____ ENGINE MODEL(S) _____ ENGINE SERIAL NO(S). _____ FLIGHT NUMBER _____	<u>TYPE OF MISSION:</u> (SELECT UP TO 2) SCHEDULED PAX CARGO UNSCHEDULED PAX FERRY FLIGHT TEST TRAINING UNKNOWN MAINT	
	<u>AIRLINE/OPERATOR:</u> OPERATOR NAME _____ OPERATOR OAG CODE _____	
<u>METEOROLOGICAL/ENVIRONMENT CONDITIONS:</u> IMC/VMC _____ CLOUD CEILING FT OR M _____ LIGHT CONDITIONS _____ DAY/NIGHT/DUSK/DAWN _____ VISIBILITY FT, M, MI _____ WIND: DIRECTION _____ VELOCITY IN KTS _____ TEMPERATURE F OR C _____ MICROBURST _____ CAT _____ WINDSHEAR _____	<u>VERTICAL TURBULENCE</u> _____ HAZE _____ HAIL _____ BIRDS _____ SNOW/SLUSH _____ SAND/ASH _____ THUN STRMS _____ LIGHTNING _____ OTHER WEATHER _____ ICE/RAIN/FOG/GUSTS _____	
<u>PHASE OF OPERATION</u>		
BOARDING	DESCENT	DEBOARDING
CARGO LOADING	APPROACH	PARKED
ENGINE START	INITIAL	REFUELING
TAXI	FINAL	INSPECTION
TAKE OFF	LANDING	TOWED
ROLL	FLARE & TOUCHDOWN	SERVICING
ROTATION	ROLL	UNKNOWN
INIT CLIMB	TOUCH AND GO	CLIMB TO CRUISE
GO AROUND	CRUISE	TAXI
DURING DIVERT		

HARDWARE INVOLVED IN INCIDENT:

----- -----
 ----- -----
 ----- -----
 ----- -----
 ----- -----
 ----- -----

ATA CODE ___ / ___

NAME _____
 MODEL _____
 MAKE _____
 LOCATION _____
 PART NUMBER _____
 TOTAL TIME _____
 TIME SINCE O/H _____
 CYCLES SINCE O/H _____
 TOTAL CYCLES _____

TYPE OF HUMAN MACHINE INTERFACE ERROR

Suggest that a coded list be developed that is similar to ATA codes

NAT. AVIATION SYSTEM (NAS): TBDFLIGHT CREW EXPERIENCE:

CAPTAIN _____
 TIME IN TYPE ACFT _____
 FIRST OFFICER
 TIME IN TYPE ACFT _____
 SECOND OFFICER
 TIME IN TYPE ACFT _____

PILOT IN COMMAND _____
 TOTAL FLYING TIME _____
 TOTAL FLYING TIME _____
 TOTAL FLYING TIME _____

DATA SOURCES:

FLIGHT CREW _____
 MAINTENANCE _____
 OPERATOR _____
 MANUFACTURER _____
 NTSB _____
 WAAS _____

ATC _____
 CAA _____
 FLT INT _____
 FLIGHT SAFETY FOUNDATION _____
 NEWS _____
 AIRCLAIMS _____
 OTHER _____

BRIEF DESCRIPTION:

Describe the event/situation. Keeping in mind the following topics, discuss those which you feel are relevant and anything else you think is important. Include what you believe really caused the problem, and what can be done to prevent a recurrence, or correct the situation. (USE ADDITIONAL PAGES IF NECESSARY)

1. CHAIN OF EVENTS

How the problem arose
 Contributing factors
 How was it discovered
 Corrective actions taken
 System configurations and operating modes
 What procedures were used
 How did you decide what to do
 What stopped the incident from becoming an accident
 Failure in Cockpit Resource Management
 Fatigue

2. HUMAN PERFORMANCE CONSIDERATIONS

Perceptions, judgements, decisions
 Factors affecting the quality of human performance
 Actions or inactions
 Lack of positional awareness
 Lack of awareness of circumstances of flight
 Incorrect selection on instrument/navaid
 Action on wrong control/instrument
 Slow/delayed action
 Omission of action/inappropriate action
 Fatigue
 State of mind
 Lack of qualification/training/experience
 Incapacitation/medical or other factors reducing crew performance
 Deliberate non-adherence to procedures

FULL NARRATIVE:ANALYST COMMENTS:

Factors Relevant to Incident
 (Each incident usually has more than one factor)

Group	Factor	No. acc.
A. Causal factors		
A.1 Aircraft systems	1.1 System failure – affecting controllability 1.2 System failure – flight deck information 1.3 System failure - other	
A.2 ATC/Ground aids	2.1 Incorrect or inadequate instruction/advice 2.2 Misunderstood/missed communication 2.3 Failure to provide separation - air 2.4 Failure to provide separation - ground 2.5 Ground aid malfunction or unavailability	
A.3 Environmental	3.1 Structural overload 3.2 Wind shear/upset/turbulence 3.3 Icing 3.4 Wake turbulence - aircraft spacing 3.5 Volcanic ash/sand/precipitation etc. 3.6 Birds 3.7 Lightning 3.8 Runway condition unknown to crew	
A.4 Crew	4.1 Lack of positional awareness - in air 4.2 Lack of positional awareness - on ground 4.3 Lack of awareness of circumstances in flight 4.4 Incorrect selection on instrument/navaid 4.5 Action on wrong control/instrument 4.6 Slow/delayed action 4.7 Omission of action/inappropriate action “Press-on-U’s” 4.9 Failure in CRM (cross-check/co-ordinate) 4.10 Poor professional judgments/airmanship 4.11 Disorientation 4.12 Fatigue 4.13 State of mind 4.14 Interaction with automation 4.15 Fast and/or high on approach 4.16 Slow and/or low on approach 4.17 Loading incorrect 4.18 Flight handling 4.19 Lack of qualification/training/experience 4.20 Incapacitation/medical or other factors reducing crew performance 4.21 Failure in look-out 4.22 Deliberate non-adherence to procedures	
A.5 Engine	5.1 Engine failure 5.2 Propeller failure 5.3 Damage due to non-containment 5.4 Fuel contamination 5.5 Engine failure simulated	
A.6 Fire	6.1 Engine fire or overheat 6.2 Fire due to aircraft systems 6.3 Fire - other cause 6.4 Post crash fire	
A.7 Maintenance/ground handling	7.1 Failure to complete due maintenance 7.2 Maintenance or repair error/oversight/inadequacy 7.3 Ground staff struck by aircraft 7.4 Loading error 7.5 SUPS - Suspected Unapproved Parts 7.6 Unapproved Parts	

Group	Factor	No. acc.
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A. *Causal factors*

A.8 Structure	8.1 Corrosion/fatigue 8.2 Overload failure 8.3 Flutter	
A.9 Infrastructure	9.1 Incorrect, inadequate or misleading information to crew 9.2 Inadequate airport support	
A.10 Design	10.1 Design shortcomings 10.2 Unapproved modification 10.3 Manufacturing defect	
A.11 Performance	11.1 Unable to maintain speed/height 11.2 Aircraft becomes uncontrollable	
A.12 Other	12.1 Caused by other aircraft 12.2 Non-adherence to cabin safety procedures	

B. *Circumstantial factors*

B.1 Aircraft systems	1.1 Non-fitment of presently available safety equipment (GPWS, TCAS, windshear warning, etc.) 1.2 Failure/inadequacy of safety equipment	
B.2 ATC/ground aids	2.1 Lack of ATC 2.2 Lack of ground aids	
B.3 Environmental	3.1 Poor visibility 3.2 Other weather 3.3 Runaway condition (ice, slippery, standing water, etc.)	
B.4 Training	4.1 Training inadequate 4.2 Presented with situation beyond training 4.3 Failure in CRM (cross-check/co-ordinate)	
B.5 Infrastructure	5.1 Incorrect/inadequate procedures 5.2 Company management failure 5.3 Inadequate regulation 5.4 Inadequate regulatory oversight	

C. *Consequences*

C.1 Controlled flight Into Terrain (CFIT) C.2 Collision with terrain/water/obstacle C.3 Mid-air collision C.4 Ground collision with other aircraft C.5 Ground collision with object/obstacle C.6 Loss of control in flight C.7 Fuel exhaustion C.8 Overrun C.9 Undershoot C.10 Structural failure C.11 Post crash fire C.12 Fire/smoke during operation C.13 Emergency evacuation difficulties C.14 Forced landing - land or water C.15 Other cause of fatality	
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D. *Unknown*

Level of confidence High Medium Low Insufficient Information

Note: Acts of terrorism and sabotage, test and military-type operations, and fatalities to third parties not caused by the aircraft or its operation are excluded.

TABLE 2: ASPECTS THAT CAN ADVERSELY AFFECT SAFETY

Design	Manufacturing	Maintenance
New Technology	Technological Obsolescence	Human Factors
Repair	Unwitting Exceedances	Configuration Management
Flight Operations	Air Traffic Control	Adverse Environment
Software	Training	Records
Regulations	Environmental Rules	Unapproved Parts
Hazardous Cargo/Stores		

Figure 1.

Accidents and Onboard Fatalities by Phase of Flight

Hull Loss and/or Fatal Accidents — Worldwide Commercial Jet Fleet — 1988 through 1997

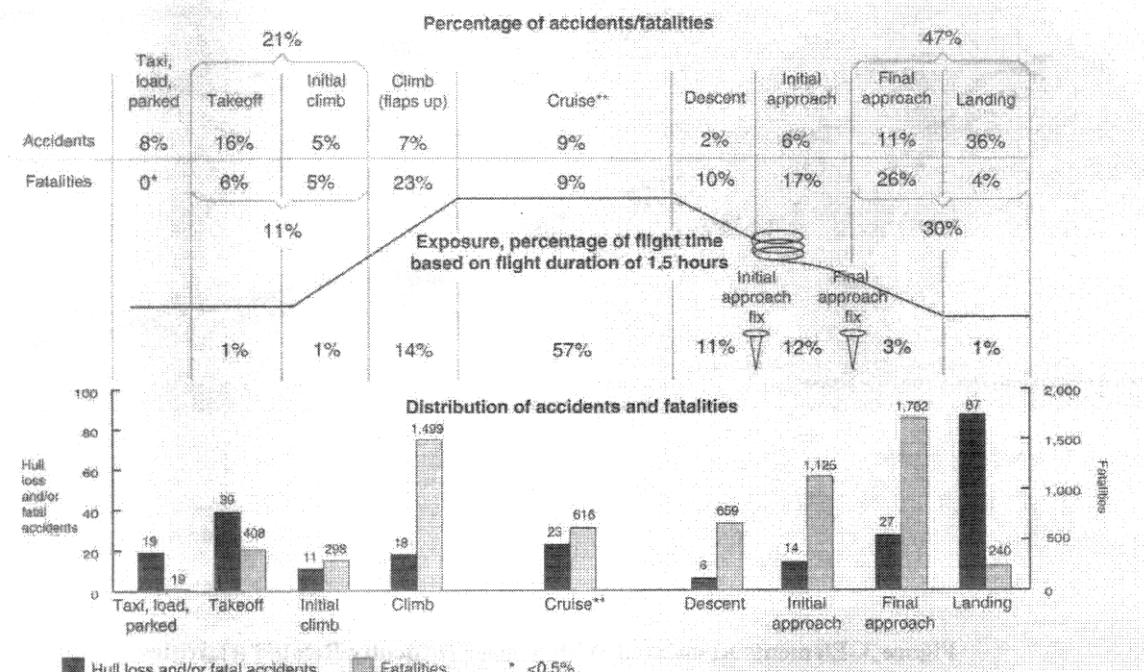


Figure 2.

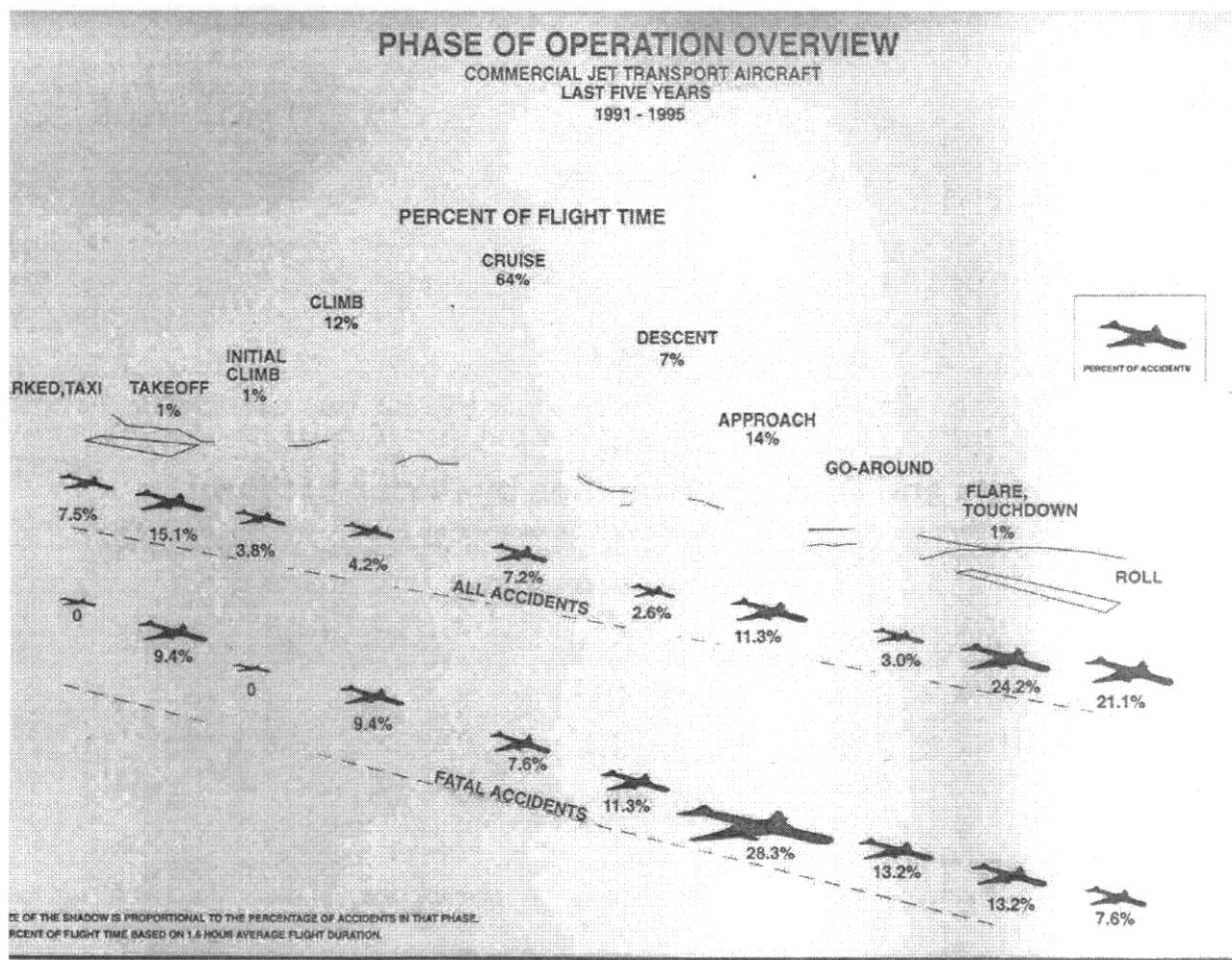
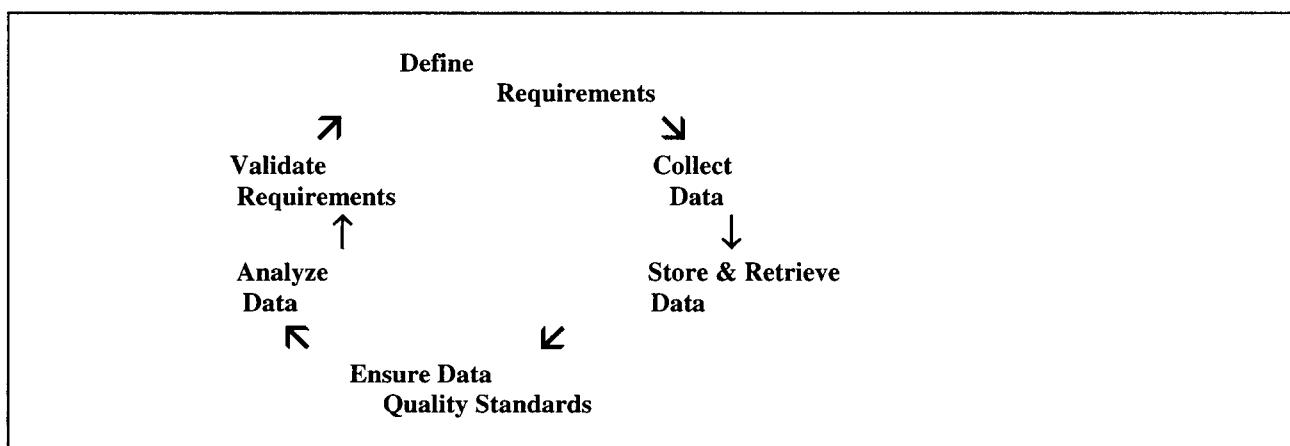


Figure 3. Elements Associated With Service Difficulty Related Activities



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13. Keywords/Descriptors	Aircraft maintenance NATO forces Aging (metallurgy) Aircraft engines Aircraft equipment Electric equipment Airborne equipment	Avionics Helicopters Structural integrity Airworthiness Service life Fatigue (materials)	Damage assessment Risk Aviation safety Noise (sound) Aging aircraft Impulse noise
14. Abstract	<p>Aging Aircraft concerns have dramatically escalated in the military community and commercial aviation during the past decade. Some models, which have already been in service for more than 40 years, will need to be retained for another two decades or longer, often serving in roles and in theatres very different from what was envisioned when they were originally designed.</p> <p>Aging Aircraft has several connotations. To name a few: technological obsolescence, the spectre of runaway maintenance costs, and safety. Moreover, spare parts, processes and tooling may no longer be available, logistic procedures may have changed and suppliers may be out of the business. Budgetary limitations and higher fleet utilisation will increase the demand to cope with aging structures and major subsystems like engines and avionics.</p> <p>Specific topics covered by this Lecture Series are:</p> <ul style="list-style-type: none"> • An operator's perspective on aging engines • Modern engine modernisation programmes • Aging electrical systems and wiring • Aging avionics • Aging helicopter-related issues • Other subsystems • Safety and service difficulty reporting 		



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